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NRL Report 3097

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925857  
**A SATELLITE AND SPACE VEHICLE PROGRAM  
FOR THE NEXT STEPS BEYOND  
THE PRESENT VANGUARD PROGRAM**

[UNCLASSIFIED TITLE]

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## CHAPTER I INTRODUCTION (SECRET)

The present Project Vanguard IGY Satellite Program is expected to be completed during the latter half of 1958. Many of the experiments to be conducted in these satellites will furnish basic information which can be used in programs for meeting urgent military and operational satellite needs. None of the Vanguard IGY satellites will, however, include any military equipment, per se.

Recent thinking, stimulated in part by the Soviet satellites, has brought into focus the fact that there are a number of very important military and operational applications of satellites which must be exploited if the Nation's relative military position is not to be seriously impaired.

Almost all of the military and operational objectives can be achieved by means of satellite payloads of the order of 300 pounds.

A program for developing and operating military and operational satellite systems will involve the following major phases:

1. Launching vehicle development,
2. Launching operations,
3. Satellite tracking,
4. Satellite orbit determination, and
5. Military and operational satellite development

↓  
This report accordingly begins with a discussion of the development and uses of military and operational satellite systems. Related uses of the satellite launching vehicle systems for lunar missions and for other purposes are also discussed. Launching vehicle systems capable of meeting the needs of the military and operational satellite programs in the immediate future are discussed ~~in Chapter 4~~. Certain aspects of the launching operation problem are also discussed ~~there~~.

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Satellite radio tracking and satellite orbit determination can be accomplished by means of the systems now established for Project Vanguard satellites. The locations of radio tracking stations for the military and operational satellite systems would probably be different in some cases, however. It now appears, for example, that it would be wise to establish radio tracking stations on American Samoa and Antarctica in order to provide the important southern hemisphere coverage from U.S. bases.

The present report deals in detail with the portions of this program which would represent the next steps beyond the present Vanguard program, namely the programs built around the Improved Vanguard and the Thor-Vanguard satellite launching vehicle combinations.

The program recommended for the period immediately beyond the present Vanguard program is summarized in Tables 1 through 5. It involves three steps:

- (1) The use of twelve improved Vanguard satellite launching vehicles for launching satellites in the 50-pound class.
- (2) The use of twenty-four satellite launching vehicles involving combinations of Thor and Vanguard stages, for larger satellites in the 300-pound class, and
- (3) A series of larger launching vehicles capable of placing payloads ranging from 1500 pounds to the order of 3 tons in orbit.

It appears that the steps beyond these will require the use of a propulsion unit with a thrust of the order of a million pounds. Accordingly, a program for developing such a propulsion system capability should be gotten underway at once.

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**TABLE 1**  
Proposed Satellite and Space Vehicle Programs for the  
Next Steps Beyond the Present Vanguard Program

| Vehicle                    | Satellite Payload (lb) |              |       | Number<br>of Vehicles | Initial<br>Launching | Purpose  |
|----------------------------|------------------------|--------------|-------|-----------------------|----------------------|--|
|                            | 90°<br>Orbit           | 30°<br>Orbit | Lunar |                       |                      |  |
| Improved Vanguard          | 35                     | 55           |       | 12                    | Nov. 58              | Navigation<br>Reconnaissance<br>Biological<br>Scientific         |
| Thor-Vanguard              | 350                    | 475          | 50    | 24                    | Jan. 59              | Moon<br>Navigation<br>Reconnaissance<br>Biological<br>Scientific |
| 1500-lb Payload<br>Vehicle |                        | 1500         |       | 6                     | Mid 59               | all above<br>+ Manned  |

**TABLE 2**  
Possible Launching Schedules

| Vehicle                 | Schedule              |   |                  |   |   |   |   |   |   |   |   |   |      |   |              |   |   |   |   |   |   |   |      |   |         |   |   |   |   |   |
|-------------------------|-----------------------|---|------------------|---|---|---|---|---|---|---|---|---|------|---|--------------|---|---|---|---|---|---|---|------|---|---------|---|---|---|---|---|
|                         | 1958                  |   | 1959             |   |   |   |   |   |   |   |   |   | 1960 |   |              |   |   |   |   |   |   |   | 1961 |   |         |   |   |   |   |   |
|                         | N                     | D | J                | F | M | A | M | J | J | A | S | O | N    | D | J            | F | M | A | M | J | J | A | S    | O | N       | D | J | F | M | A |
| Improved Vanguard       | (Cape Canav-<br>eral) |   |                  |   |   |   |   |   |   |   |   |   |      |   |              |   |   |   |   |   |   |   |      |   |         |   |   |   |   |   |
|                         |                       |   |                  |   |   |   |   |   |   |   |   |   |      |   | (Camp Cooke) |   |   |   |   |   |   |   |      |   |         |   |   |   |   |   |
|                         | 1                     | 1 | 1                | 1 |   |   |   |   |   |   |   |   |      |   | 1            | 1 | 1 | 1 | 1 | 1 | 1 | 1 | 1    | 1 | 1       |   |   |   |   |   |
| Thor-Vanguard           |                       |   | (Cape Canaveral) |   |   |   |   |   |   |   |   |   |      |   | (Camp Cooke) |   |   |   |   |   |   |   |      |   |         |   |   |   |   |   |
|                         |                       |   | 1                | 1 | 1 | 1 | 1 | 1 | 1 | 1 | 1 | 1 | 1    | 1 | 1            | 1 | 1 | 1 | 1 | 1 | 1 | 1 | 1    | 1 | 1       | 1 | 1 | 1 | 1 | 1 |
| 1500-lb Payload Vehicle |                       |   |                  |   |   |   |   |   |   |   |   |   |      |   |              |   |   |   |   |   |   |   |      |   |         |   |   |   |   |   |
|                         |                       |   |                  |   |   |   |   |   |   |   |   |   |      |   | 1 1          |   |   |   |   |   |   |   |      |   | 1 1 1 1 |   |   |   |   |   |

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TABLE 3  
Proposed Uses of Improved Vanguard Vehicles

|  |    |
|--|----|
| 12 Satellites                                  |    |
| Nov. 1958 to Feb. 1959 - Cape Canaveral        |    |
| May 1959 to Dec. 1959 - Camp Cooke             |    |
| <u>Cape Canaveral</u>                          |    |
| Biological and Scientific                      | 2  |
| Satellite Development and Scientific           | 2  |
| <u>Camp Cooke</u>                              |    |
| Electronic Intelligence                        | 2  |
| Navigation and Scientific                      | 2  |
| Weather Studies and Scientific                 | 1  |
| Atomic Test Surveillance                       | 2  |
| Missile Test Surveillance (Infrared & Optical) | 1  |
| Total  | 12 |

TABLE 4  
Proposed Uses of Thor-Vanguard Vehicles

| <u>Purpose</u>                                | Thor-Vanguard<br>8 Satellites            | Improved Thor-Vanguard<br>16 Satellites |
|---|--|---|
|   | Jan. 1959 to Oct. 1959<br>Cape Canaveral | Jan. 1960 to Apr. 1961<br>Camp Cooke    |
| Moon  | 1  | 2                                       |
| Biological                                    | 1  | 2                                       |
| Satellite Development                         | 1  | 2                                       |
| Recovery, etc.                                |  |   |
| Weather and Scientific                        | 1  | 2                                       |
| Studies                                       |  |   |
| Reconnaissance                                | 2  | 4                                       |
| Atomic Test                                   |  |   |
| Missile Test                                  |  |   |
| TV  |  |   |
| Navigation, Communication                     | 1  | 2                                       |
| Reconnaissance (Dark)                         |  |   |
| Radioactivity                                 |  |   |
| Electronic                                    |  |   |
| TV  |  |   |
| Polar Navigation, Communications              | 1  | 2                                       |
| Weather and Scientific Studies<br>(announced) |  |   |
| Total   | 8  | 16                                      |

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TABLE 5  
Proposed Uses of 1500-Pound Payload Vehicles

| 1500-lb Payload Vehicle<br>6 Satellites<br>Mid 1959 and 1960 |   |
|--|---|
| <u>Cape Canaveral or Camp Cooke</u>                          |   |
| Biological & Recovery  | 2 |
| Moon   | 2 |
| Reconnaissance   | 2 |

The following individuals made major contributions to the sections indicated, under the guidance of Project Vanguard Director Dr. John P. Hagen.

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|                         |             |
|-------------------------|-------------|
| Neutron Reconnaissance: | W. R. Faust |
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|                         |             |
|-------------------------|-------------|
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|---|---------------------------------|
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The material in this report was coordinated and compiled by Dr. J. W. Siry and  
Mr. K. Squires.

## CHAPTER 2

# MILITARY, OPERATIONAL, AND SCIENTIFIC SATELLITE SYSTEMS

### 2.1 MILITARY AND OPERATIONAL SYSTEMS

#### 2.1.1 INTRODUCTION (SECRET)

This chapter contains discussions of various types of satellite systems. The different uses are discussed in terms of military and operational applications of satellites, supporting researches, scientific research, and satellite development. This division is somewhat arbitrary; certain of its developments might logically be included in more than one category. The arrangement adopted here does, however, call attention to several fundamental problems of satellite technology.

The time estimates made in the following sections are, in most cases, made without the benefit of detailed analyses. They are in many cases, however, made by individuals or teams which have had long experience, approximately a decade in many cases, in the interrelated fields of which they speak. This is true of all of the major programs proposed in the sections on military applications and supporting research, and of most of the scientific research programs proposed.

This means, for example, that the nuclear reconnaissance equipments proposed could be tested and checked out at the AEC's proving grounds in the atomic weapons test series by the NRL groups which normally participate in these tests and which formulated the proposals of this report. Similar observations apply to the navigation, electronic intelligence, infrared, and optical programs proposed here. Each of the estimates is made subject to the assignment of appropriate priorities.

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### 2.1.2 NAVIGATION SYSTEMS

#### 2.1.2.1 Principles of Radio Navigation

##### 2.1.2.1.1 The Theory of Radio Navigation by Means of Artificial Earth Satellites (Secret)

The theory of the determination of the position of an observer by means of zenith distance observations of artificial satellites is presented in this section. The principal advantage of radio navigation by means of satellites lies in the fact that it enables one to take advantage of present-day techniques for measuring time with high accuracy. Accurate measurements of time can, to a certain degree, compensate for less accurate measurements of position.

It is assumed in the following discussion that the latitudes,  $\phi_i$ , longitudes,  $\lambda_i$ , and heights,  $h_i$ , of the satellite, for  $i = 1, 2, \dots$  are known from ephemerides. These quantities for  $i = 1, 2$  are indicated in Fig. 1. In that figure,  $R$  denotes an appropriate value of the earth's radius,  $O$  denotes the observer's location, the subscript  $o$  denotes quantities associated with the observer's position. It follows that from consideration of the triangle  $C, S_1, O$  that

$$\sin a_1 = \frac{R}{R + h_1} \sin z_1.$$

$$\ell_1 = z_1 - a_1.$$

It is seen with the aid of Fig. 2, that the following relations hold:

$$\cos \epsilon_1 = \frac{\cos \phi_2 - \cos \phi_1 \cos \Delta P_1}{\sin \phi_1 \sin \Delta P_1},$$

$$\cos \gamma_1 = \frac{\sin \phi_2 - \sin \phi_1 \cos \Delta P_1}{\cos \phi_1 \sin \Delta P_1},$$

$$\gamma_1 = \gamma_1 - \epsilon_1.$$

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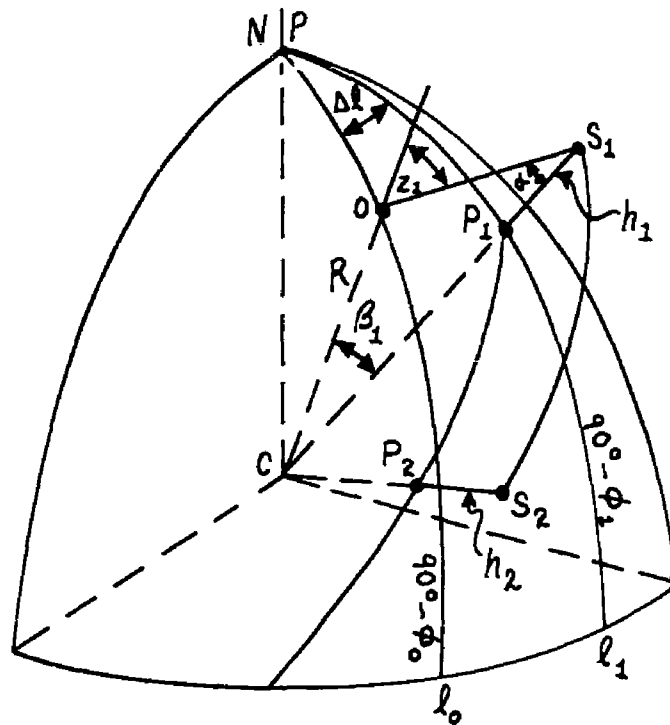


Fig. 1 - Three-dimensional geometry of navigation problem

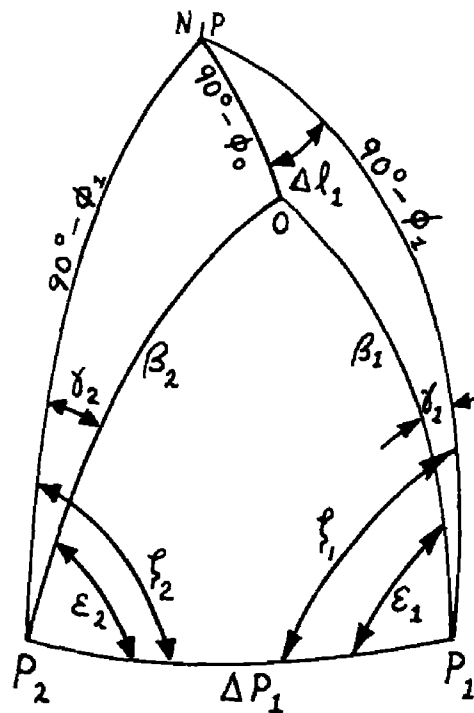


Fig. 2 - Earth-surface geometry of navigation problem

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It follows from consideration of the triangle  $\overline{NP}, O, P_1$ , that the observer's latitude is given by:

$$\sin \phi_o = \sin \phi_1 \cos \delta_1 + \cos \phi_1 \sin \delta_1 \cos \gamma_1.$$

It then follows that

$$\cos \Delta \lambda_1 = \frac{\cos \delta_1 - \sin \phi_o \sin \phi_1}{\cos \phi_o \cos \phi_1}.$$

The observer's longitude can then be determined using this result together with the satellite's longitude,  $\lambda_1$ , which can be obtained from the ephemerides.

### 2.1.2.1.2 The Navigational Fix Derived From Sumner Lines of Position Based on Satellite Observations (Secret)

A method of finding an observer's position by computational methods associated with the determination of Sumner lines of position is presented in this section.

When a satellite's latitude  $\phi_s$ , longitude  $\lambda_s$ , and height  $h$  are known, the Sumner line of position corresponding to an observed elevation angle  $\theta$  may be determined as follows.

If  $h$  is given in statute miles, the satellite's ground range  $d^o$  on a spherical earth is given by

$$d^o = \cos^{-1} \left[ \frac{3960}{3960 + h} \right] \cos \theta = \theta. \quad (1)$$

Assuming a convenient latitude  $\phi_o$  of the observer within a degree of so of the true, the corresponding satellite-observer difference in longitude  $(\lambda_o - \lambda_s)$  is given exactly by

$$(\lambda_o - \lambda_s) = \cos^{-1} \left( \frac{\cos d^o - \sin \phi_o \sin \phi_s}{\cos \phi_o \cos \phi_s} \right). \quad (2)$$

Also, the satellite's azimuth  $A$  is given by

$$A = \sin^{-1} [\sin(\lambda_o - \lambda_s) \cos \phi_s \cos d^o]. \quad (3)$$

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Solutions to Eqs. (2) and (3) are tabulated in Hydrographic Office Publication No. 214, to an accuracy of about 0.1° without interpolation. Quick approximate graphical solutions to Eqs. (1), (2), and (3) may be found using nomograms published in Annals of the International Geophysical Year, Vol. IV, Rockets and Satellites, L. V. Berkner, Editor; London Pergamon Press, 1957.

The resulting Sumner line of position, which may be assumed straight on a plane earth, is then drawn through the assumed position of the observer, perpendicular to the azimuth line. The intersection of two Sumner lines of position gives a navigational fix of the observer. As a check and for increased accuracy, usually three or more lines of position are determined and the mean of intersection points is taken as the fix.

The error  $\Delta d^0$  in the parallel position of the Sumner line is related to the elevation error  $\Delta \epsilon$  by

$$\Delta d^0 = \left[ \frac{\left( \frac{3960}{3960 + h} \sin \epsilon \right)}{\sqrt{1 - \frac{3960}{3960 + h} \cos^2 \epsilon}} - 1 \right] \Delta \epsilon \quad (4)$$

This has an obvious maximum of  $\Delta d^0 = -\Delta \epsilon$  at  $\epsilon = 0$  for any  $h$ , which maximum is also constant for all values of  $\epsilon$  when  $h$  is very large. For smaller values of  $h$ , the minimum  $\Delta d^0$  occurs for  $\epsilon = 90^\circ$  and is given by

$$\Delta d^0 = \left( \frac{3960}{3960 + h} - 1 \right) \Delta \epsilon \quad (5)$$

Thus, for instance, for  $h = 396$  miles and an elevation error  $\Delta \epsilon = 1/3$  minute of arc, the minimum,  $\Delta d^0$ , for observation at the zenith would be -0.033 nautical mile or about 200 feet. This minimum error would increase approximately in proportion to  $h$ , while the maximum error in  $\Delta d^0$  of -0.333 nautical mile or 2000 feet would be the same for any value of  $h$ . Also, for sightings near the zenith, the assumed position of the observer

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must be moved closer to the true, in order that the assumption of straight Sumner lines may remain a valid approximation.

For an assumed latitude of the observer within a degree or so of the true, the effect of the corresponding error in azimuth is relatively negligible.

The error in longitude of a fix contributed by any Sumner line of position is  $\Delta \ell = \Delta \ell^0 \cos \lambda \sin A$ , and in latitude is  $\Delta \lambda = \Delta \ell^0 \cos A$ , where  $A$  is the azimuth. Hence the probable error in longitude of a fix derived from  $n$  Sumner lines of position is given by

$$\Delta \ell = 0.6745 \sqrt{\frac{\sum_{k=1}^n \Delta \ell_k^0 \sin A_k \cos \lambda^2}{n(n-1)}} \quad (6)$$

The probable error in latitude of a fix derived from  $n$  Sumner lines of position is

$$\Delta \lambda = 0.6745 \sqrt{\frac{\sum_{k=1}^n \Delta \ell_k^0 \cos A_k}{n(n-1)}} \quad (7)$$

Thus for ten observations near the zenith for which the error  $\Delta \ell^0 = 200$  feet as above, the corresponding error  $\Delta \lambda$  of the fix would be about 45 feet.

### 2.1.2.2 Radio Navigation System

#### 2.1.2.2.1 A 300-Mc Navigation System (Secret)

An observable body in the heavens whose time and position were precisely predictable would be a direct aid to navigation. The establishment of a satellite in an orbit sufficiently high to preclude rapid loss of its energy to the upper atmosphere and thus to assure a reasonable lifetime, would provide such a body. A radio transmitter in this satellite would provide a means of tracking it accurately and measuring its times of passage, as is presently contemplated with the Minitrack system. For application to Naval problems, a two-phase program is proposed:

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For immediate application by surface ships the present type of Minitrack system, operated at a frequency three to four times higher than at present, would provide tracking accuracies many times greater than those now obtainable. The use of a satellite transmitter with a higher power output and a solar-cell power supply would permit the use on shipboard of relatively small antenna arrays, of the order of 5 by 5 feet, on baselines of not more than 50 feet, with an unlimited operating lifetime. If the satellite altitude were 500 miles this system would permit ship location to within less than 1/4 nautical mile. The system, employing transistorized satellite transmitters, could be produced using available techniques and components. It would weigh about 24 pounds and could be applied to the location of submarines without surfacing, since it could be used to chart specific ocean bottom features and to locate sono-beacons as future submarine position-finding points. Such a system could be readied in about a year at a cost of about a million dollars.

### 2.1.2.2.2 An X-Band Navigation System (Secret)

For later application to submarines, the use of a modified Minitrack system at or near the X band (10,000 Mc) is proposed. This proposal is similar to the proposal of Reference (1), which includes the use of the moon as a reflection target for alignment of the X-band receivers, and also as an auxiliary navigation source. Both of these systems would provide data outputs in the form desired for immediate computation or navigation. The present Minitrack system data outputs are available in both digital and analog form; either form can easily be converted in real time to any other form required, within the accuracy of the basic system.

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### 2.1.3 RECONNAISSANCE SYSTEMS

#### 2.1.3.1 Nuclear Weapons Test Reconnaissance

##### 2.1.3.1.1 Thermal Radiation Reconnaissance (Secret-RD)

The following study is limited to techniques which can be used within two years of this date and which involve satellites in the 50-pound or 300-pound class.

The simplest question on nuclear weapons testing which a satellite can answer is whether a weapon has been detonated within its field of view.

The second question that might be answered is the time of firing.

The third question is the location of the detonation. Detonation location assumes a stabilized satellite whose reference frame is in known orientation to that of the earth. The error associated with the weapon test detection equipment, per se, is of the order of 1,000 yards.

The fourth question is the yield of the burst. The most promising technique for measuring the yield of the burst is to take the record of radiant power as a function of time and determine the yield from the time of the minimum of such a curve, or the time of the second maximum. The yield of a surface burst can be determined with an accuracy of a few percent. If a stabilized satellite is used, such measurements can be made by day or by night.

The fifth question that might be answered is the height of the burst. Several techniques may be used for this. The most promising is the same as that described for the yield measurement. The shape of the power curve from a detonation is dependent on the altitude of the burst. Between a surface burst and a 100,000-foot burst the amount of energy in the first maximum differs by at least a factor of ten, and the time of the second maximum in the 100,000-foot burst decreases to approximately 60 percent of the surface time, while the minimum time is approximately unchanged. Other techniques for determining the burst altitude include observation of the light from the detonation separately in the visible and the ultraviolet regions. If ultraviolet light is observed it may be concluded

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that the detonation took place above the ozone layer which is at approximately 30 kilometers. Similar techniques utilizing the attenuation of gamma rays and neutrons by air have been considered. The power curve method is considered most certain.

For detonations with high yield-to-mass ratios, the power curve is modified in a recognizable way during the development of the first maximum. It is believed that if a clean power curve were obtained, some information on the yield-to-mass ratio would be available.

The measurement of radial power as a function of time is basic in connection with many of the questions discussed above. This measurement is discussed in more detail in the following paragraphs.

During the detonation of a nuclear weapon, the light history is very sensitive to the weapon's yield and to the altitude of the burst, and can serve as a key to some construction characteristics. Typical radial phenomena as a function of time, interpreted in terms of the light output, are shown in Fig. 3. The three initial flashes of light diverging from the point of detonation at zero time are due to gamma rays, 14-mev neutrons, and thermal neutrons. In a conventional shot, the fireball is seen to grow by radiative diffusion processes starting essentially at zero time and growing with great rapidity to a radius of approximately

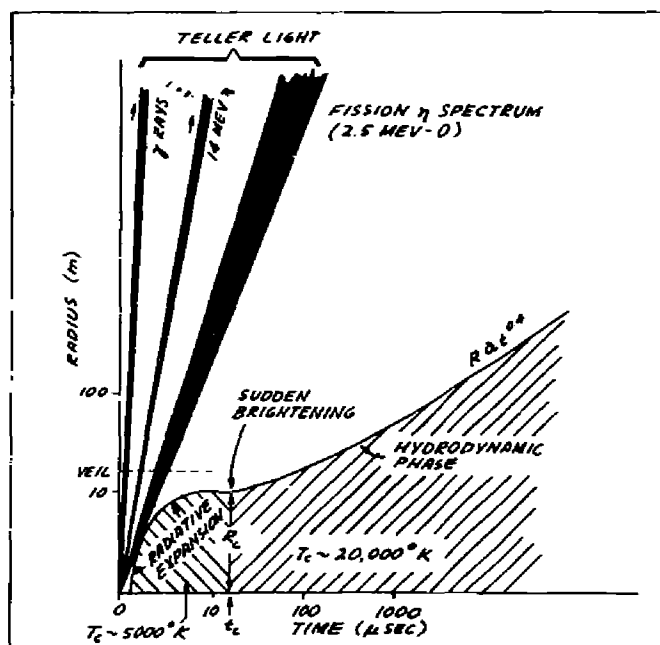


Fig. 3 - Radial phenomena of a nuclear detonation

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10 meters, whereafter the growth rate decreases. At a time of approximately 20-30  $\mu\text{sec}$  the hydrogenamic expansion gets underway and thereafter the fireball growth is due to shock running in the air.

When the detonation has an extremely high yield-to-mass ratio, the case temperature can be so great that x-rays from the case render the air around the point of detonation opaque and slightly luminous to radii of the order of 20 meters. All phenomena occurring within this radius are blacked out by the absorbing air and the whole appearance of the fireball development up to times, for kiloton weapons, of the order of 150  $\mu\text{sec}$  is modified. For megaton weapons, this time may be as great as 25 milliseconds and can wipe out the entire first maximum. The light history evaluated in terms of intensity as a function of time is shown in Fig. 4. The figure is drawn for a 20-kiloton detonation and the changes in the general shape of the curve with increased yield are indicated. Note that  $t_0$  represents the light due to air fluorescence from the gamma and neutron irradiation of the air.  $t_a$  represents a transition from the radiative expansion phase to the hydrodynamic, and this time is sensitive to the yield and the mass of the weapon fired.  $t_b$  represents the time when the fireball burns through the opaque air generated by x-rays for those cases where

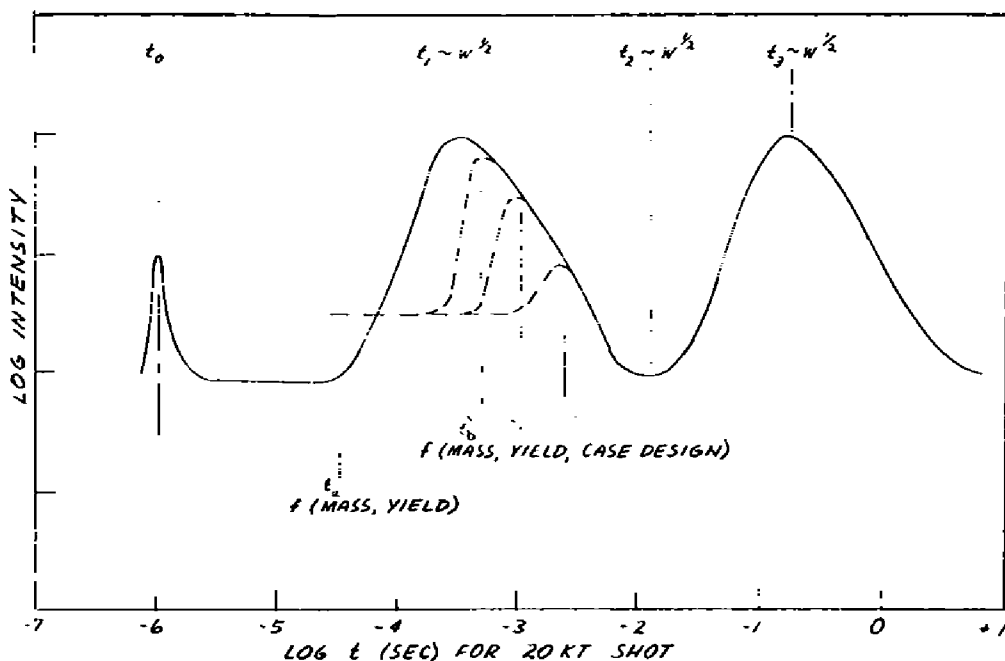


Fig. 4 - Light history of a 20-kiloton nuclear detonation

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the yield-to-mass ratio is favorable, and this time is sensitive to the yield-to-mass ratio and to the case design of the weapon. The times of the maxima and the minimum,  $t_1$ ,  $t_2$ , and  $t_3$ , vary proportionally to the square root of the yield. The time  $t_2$  is essentially insensitive to the altitude of the burst. The time  $t_3$  decreases as the altitude of the burst increases and the ratio  $t_3/t_2$  can be used as an index to the altitude of the burst. The ratio of the area under the first maximum to the area under the second maximum increases in a sensitive way with altitude.

Knowledge of the curve of radiant power as a function of time, therefore, can be interpreted to give the yield and altitude of the weapon fired, and diagnostic information about the design of the weapon.

To illustrate the type of instrumentation necessary for obtaining yield and height-of-burst information, the design of a possible satellite will be outlined in which the radiant power as a function of time could be recorded with time resolution of  $10^{-4}$  seconds, and this information could be broadcast on command of a ground station.

The reflected sunlight from the earth's surface seen by the satellite will be roughly 100 times the peak flux from a 20-kiloton detonation 1,000 miles away. Therefore, the detector must discriminate the detonation light from light due to the large low-brightness earth surface. This is done by a light-modulating system described below. Geometrically, the satellite takes the form of a cylinder, which must be launched with its spin orientation tangent to the surface of the earth in the region of interest (Figs. 5 and 6).

Located centrally inside the satellite is a wide-field-of-view photocell capable of "seeing" to the horizon of the earth in all directions. The outer shell of the cylinder is of a banded window construction, i.e., consists of alternate bands of transparent and opaque material, perhaps 300 in number. Light received from any small source within the detection sensitivity of the photocell is then chopped at 300 times the spin frequency of the cylinder, whereas light from an extended source (the earth's albedo) is essentially unmodulated. With a spin frequency of about 67 cps a chopping rate of 20,000 pps is

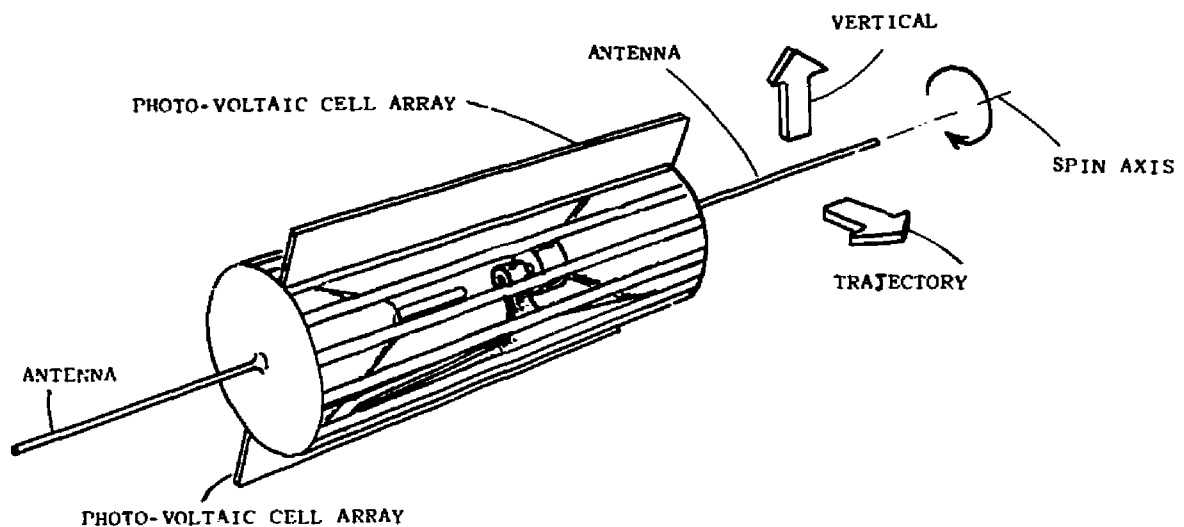


Fig. 5 - Light modulation technique of thermal radiation reconnaissance satellite

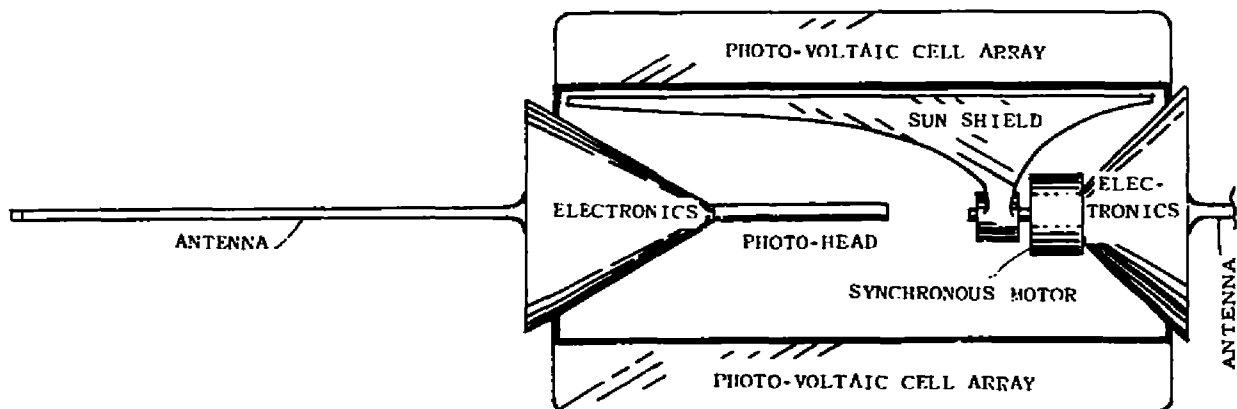


Fig. 6 - Internal arrangement of thermal radiation reconnaissance satellite

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accomplished, leading to a time resolution capability of about  $10^{-4}$  seconds. By standard techniques the modulated photocell signal is amplified and stored on a memory device in a transponder unit (Fig. 7). When the transponder is keyed by a suitable signal, the stored information is played back and transmitted to the ground receiving station. Upon completion of a preset number of playbacks the stored information is wiped out and the memory device reset for a new run.

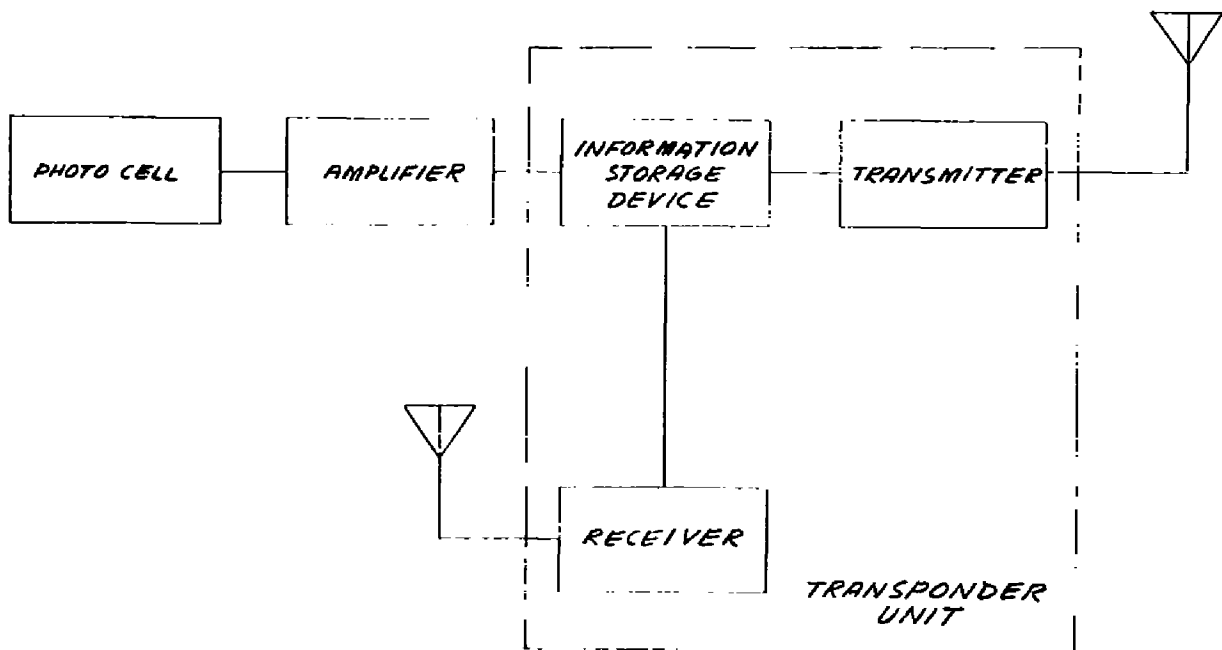


Fig. 7 - Data system for thermal radiation reconnaissance satellite

In this system the sun is a "small" source and consequently produces a modulated output signal in addition to saturating the photocell-amplifier combination. The peak light flux incident on the satellite from a 20-kiloton detonation 1,000 miles away will be of the order of  $10^{-3}$  times the steady solar illumination. Consequently, the photocell's field of view must be shielded from direct sunlight. This could be achieved by using fixed shields in a satellite launched to behave like the moon, that is, to keep one face constantly presented to the earth. Since this seems quite difficult, an alternate system is described here. In this system the shielding is accomplished by a stabilized sunshield inside the cylinder. To stabilize and orient the sunshield, a near-frictionless (because of the lack of any weight) synchronous motor with the sunshield connected to the armature is used.

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The synchronous motor is powered by solar cells mounted externally to the cylinder. Two rows of cells are mounted in such a fashion on the cylinder that their electrical output (converted from sunlight) is modulated by the rotation of the satellite and the phase is always such as to orient the sunshield in the proper direction by means of the synchronous motor.

In the absence of the sun, the sunshield will drift randomly. One must then either accept a small probability of obscuration of a nuclear signal or provide thermocouple detection of the earth's radiation, giving the possibility of lock-in of the sunshield away from the direction of the earth.

A somewhat similar system can be used to obtain the diagnostic information included in the interval from  $10^{-6}$  to  $10^{-3}$  seconds, after a detonation. This is not included in the proposal outlined here. However, some aspects of the system are obvious, including: elimination of the light modulator, use of a photocell capacity-coupled to its preamplifier, am or fm modulation of the transmitter carrier to convey the essentially video photocell signal, and preservation of the sunshield described above.

It is estimated that the development of such a satellite would take approximately 1-1/2 years. Further time could be required for check-outs in the atomic weapons test series and in flights. It is estimated that a satellite of the type described above could be built to weigh about 300 pounds. If solar power supplies are available it may be possible to package the units in a 40-pound satellite.

### 2.1.3.1.2 Neutron Reconnaissance (Secret RD)

A satellite may serve as an instrumentation station for the long-range detection and diagnostic interpretation of a nuclear explosion in free space. The absence of the attenuating effects of air permits extremely long ranges to be achieved with relatively small effort.

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Consider a satellite at a distance of 10,000 miles ( $1.6 \times 10^9$  cm) from a 100-kiloton fusion explosion. If it is assumed that 50 percent of the yield goes into 14-mev neutrons then the number of neutrons incident per unit area at 10,000 miles is

$$n = \frac{100}{4\pi (1.6 \times 10^9)^2} \times \frac{4.18 \times 10^{12} \text{ joules}}{\text{kiloton}} \times \frac{10^{13} \text{ mev}}{1.6 \text{ joule}} \times \frac{1}{2} \times \frac{1}{14 \text{ mev}}$$

$$2.9 \times 10^6 \frac{\text{neutrons}}{\text{cm}^2} \quad (1)$$

This flux is easily detectable and actually is sufficiently high that "time of flight" observations (the Tenex experiment) may be made of the neutrons to determine the neutron spectrum which characterizes the type of weapon.

If  $d$  is the distance between weapon and detector, then neutrons of energy  $E$  make the transit within a time

$$t = \frac{d}{v} = \frac{d}{\sqrt{\frac{2E}{m}}} = \frac{1.6 \times 10^9}{5.3 \times 10^9} = 0.3 \text{ sec} \quad (2)$$

However, all neutrons released by the weapon do not have the same energy but are spread over an interval  $\Delta E$ , so that there is a corresponding spread  $\Delta t$  in the arrival time at the detector:

$$\frac{\Delta t}{t} = \frac{\Delta E}{2E} \quad (3)$$

From weapon data it is found that the spread in neutron energy around 14 mev is about 300 kev, although this figure varies from weapon to weapon. From Eq. (3) the time spread in neutron arrival time is found to be

$$\Delta t = (0.3) \left( \frac{0.3}{2 \times 14} \right) = 3.3 \times 10^{-3} \text{ sec} \quad (4)$$

If it is assumed that all the neutrons are distributed over the time interval  $\Delta t$  then the neutron flux is the order of  $10^9$  neutron/cm<sup>2</sup> per second.

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The neutrons may be detected by means of a plastic scintillator, photomultiplier, and amplifiers. Signals from the amplifiers can be telemetered back to earth for oscillographic recording. It is estimated that the neutron flux calculated above, incident upon a 1,000-cm<sup>2</sup> scintillator, would produce a 1-milliampere peak current at the photomultiplier collector, or of the order of 10 volts across a 10,000-ohm load. A signal of this magnitude is easily amplified to sufficiently high levels for telemetering and oscillographic recording. The recordings will indicate the energy of the neutrons (from time of flight) and the temperature at which the nuclear reaction occurred (from time smearing). Thus, several important weapon characteristics can be obtained from satellite observations at great distances from the explosions.

It is estimated that the equipment for this experiment could be readied in about a year. A 300-pound satellite would be more than adequate. The experiment may be compatible with a 40-pound satellite, if solar power supplies are available.

### 2.1.3.1.3 Radioactivity Reconnaissance (Secret RD)

An item of deep concern to the AEC and to the services, because of its restrictive effect on weapons testing, is the accumulation in the upper air of radioactive decay products from the detonation of nuclear weapons. The AEC estimates this concentration, and limits weapons testing to keep it below an arbitrary safety level.

It is desirable to measure this concentration in the upper air for radioactivity survey purposes. Such measurements made on a day-to-day (or orbit-to-orbit) basis could be used for reconnaissance of weapons testing by other nations. These measurements would be extremely important in connection with the detection of nuclear weapons tests conducted by other nations at very high altitudes, i.e., at and above 100,000 feet. Nuclear weapons tests at these altitudes may prove to be exceedingly difficult to detect by other means.

The satellite used for this purpose should be in a polar orbit, in order to scan both hemispheres, and at as low an altitude as is compatible with the total time desired for the measurement. Since fairly long periods of measurement may be required, a 300-pound

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satellite seems desirable, in order that it may accommodate a large supply of power. The use of solar cells may make it possible to conduct these measurements in 40-pound satellites.

The satellite instrumentation (Fig. 8) should be designed to make at least four measurements per revolution. Thus, if the period were 90 minutes there would be 32 measurements to be recorded in a twelve-hour period. If the satellite can be received on either north-south or south-north transits, this is adequate for complete data. If the quarter-period measurement cycle were 22.5 minutes, the instrument program would be, generally, 21.5 minutes for sample collection, and 1 minute for data recording. During the 21.5-minute collection time the only power consumption would be by the timing clock.

The data recording cycle consists of the following actions:

1. The radioactive sample is transferred to the detector which reads a counting rate proportional to the activity on the sample. (The sample transfer activates the recording equipment.)
2. The counting rate is transferred to a "hold" circuit.
3. The voltage, proportional to the counting rate, is applied to a subcarrier oscillator (VCO).
4. VCO frequency from before to after application of the signal voltage is recorded on tape.
5. At completion of a recording cycle, the timer stops the recording equipment, and collection of another sample begins.

When the satellite passes home station, it performs the read-out cycle on command. This cycle consists of the following actions:

1. Transmitter warm-up.
2. Second command starts playback cycle, perhaps with recorder running with playback speed 8-10 times the recording speed.
3. Tape signals modulate telemetering transmitter.

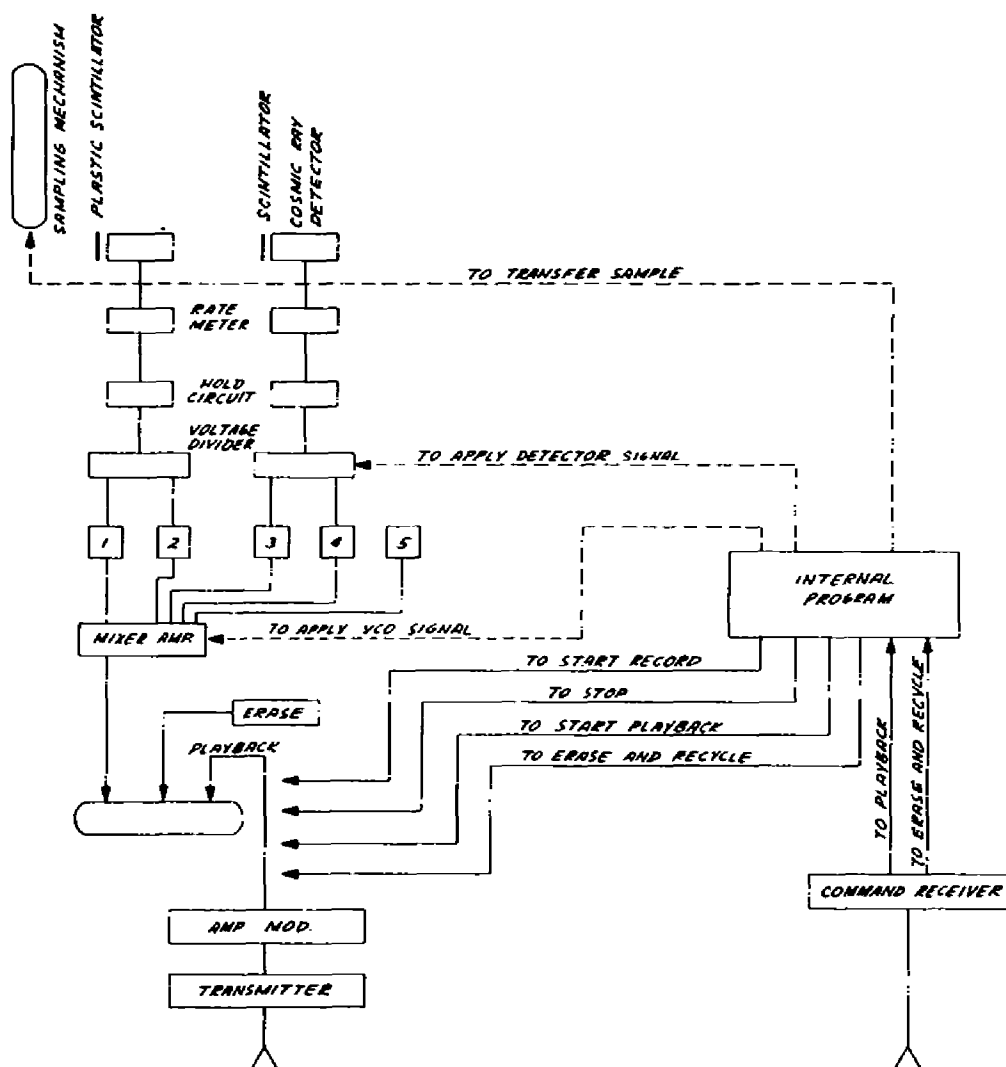


Fig. 8 - Instrumentation system for neutron reconnaissance satellite

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4. Ground station records telemeter video output on standard tape, perhaps for playback at reduced speed.

5. Final command erases tape and recycles.

The techniques of radioactive particle collection have been worked out for applications at sea level. The NRL air particle sampler for atomic submarines is an example of such a detector. Particle samples have been collected from rockets. A combination of these techniques could be used to collect the sample. Sample collection must be made at ambient pressure, and if the electronics require pressurization the count must be made at ambient pressure. Thus, the sampling process must be outside the pressurized volume. Since the radioactive particles probably will be charged because of  $\beta$  and  $\alpha$  emissions, they will be attracted or repelled, depending on the satellite's charge relative to the particles. For this reason, effective collection will require a method of adjusting satellite potential, and possibly a re-entrant volume in which an electric field can be maintained to aid collection of the particles.

The calibration of the collection system presents a unique problem. It will probably be necessary to calibrate in the laboratory at the lowest feasible pressures in large-volume vacuum systems where the particle velocity is small relative to those associated with the satellites.

In order to reduce background radiation counts, a thin plastic fluor should be used as a detector of the  $\beta$ -activity of the particles. The scintillations would be detected by a miniature photomultiplier. The counting rate meter following the photomultiplier should have a fairly long time constant in order to handle small rates. A second identical detector system with no samples would be required to measure the cosmic ray background rate during the time the sample is counted.

Since long recording times and short read-out times are required, a two-speed or an intermittent tape system is applicable. Electronic binary storage is also a possibility, but the complexity might lead to greater weight than that of the recorder. If the tape system

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is found to be suitable, the count rate signals would be used to modulate the lowest RDB subcarrier oscillators (VCO). The first five RDB bands lie below 1500 kc. A tape speed of 1 inch per second is sufficient to record these frequencies. The lowest-frequency subcarrier (400 cps) has a rise-time limitation of about 0.1 second, so that about one second must be allowed to record its information. If the first five VCO's are used, one VCO can serve as a frequency standard (1,300 cps) and can be used to compensate tape speed fluctuations. The other four can be used to extend the dynamic range of the system by using two or more channels with different sensitivities for each detector. Additional channels can be provided by duplicating the VCO's, using multitrack tape and commutating the tracks at playback.

At a tape speed of one inch per second about one inch of tape is used for each measurement. Thus, about a 50-inch tape loop is required if read-out can be effected once every 12 hours.

The tape drive mechanism will be designed to start and stop in a small fraction of a second (less than 0.1 second).

The program control will start the tape, apply CO's signals, apply signal to VCO's at proper time intervals, and then stop the tape recorder.

At a command signal from ground station, the tape will be played back continuously to the transmitter in standard fashion. The playback can be repeated several times and restored to record condition by another command. The tape is erased by the last command. The ground station might require a tracking antenna in order to maintain the telemeter link for a long enough period to get several playbacks. Alternatively, playback can be effected at 10 times the record speed, with the receiving tape operated at the same ratio so that standard discriminator data reduction can be used.

The internal program will have to be set in terms of a predetermined period for the satellite. If the satellite period varies from that planned, it should be possible to correlate data and satellite position after the fact by observing the satellite transits at the home station.

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The internal program will be controlled by a sequence cam on a clock such that the cam makes one revolution in each quarter of an orbital period. This cycle runs continuously and is pre-empted by command signals from the ground station.

Assuming availability of manpower, preparation would require about a year, with possibilities of another year required before routinely successful flights could be made.

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### 2.1.3.2 Electronic Intelligence Reconnaissance Satellite (Secret)

#### 2.1.3.2.1 An Initial Electronic Intelligence Reconnaissance Satellite (Secret)

To satisfy an urgent operational requirement for electronic intelligence, a satellite vehicle appears to be the only satisfactory means available in the near future. A particularly important and long-sought target is out of range for ground-based sites and conventional airborne platforms.

A satellite payload capability of the order of five to ten pounds apparently will be sufficient to carry the equipment required to assure detection of a prime target. This payload is based on an operating period of three weeks.

The reception of the intelligence can be accomplished at existing sites with a minimum of cost.

To be effective the satellite must, of course, have an orbit that passes over the target or at least comes within line-of-sight of it. The target of prime importance is in the Moscow complex. Therefore, the orbit must approach a latitude of 45 degrees.

Present capabilities of the Cape Canaveral range may deter the launching of vehicles on orbits of the required inclination, although 45-degree orbits might be achievable if the need is sufficiently great. The planned new test range in the western U. S. is to provide the capability of launching at any inclination.

The orbits of 1957 Alpha and Beta are considered ideal for reconnaissance satellites since they cover practically all of the inhabited world. The reception of the intelligence transmission from the proposed satellite could be accomplished from several sites. A most important fact is: these receiving sites are already existent and would require a minimum of additional equipment and manpower.

The proposed system is as simple as possible in order to be reliable, economical, and expedient. It consists only of proved circuits that are now being employed successfully and require no research work.

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The system utilizes a microwave antenna, a bandpass filter, a crystal detector, a simple video amplifier, a pulse stretcher circuit, a modulator, a tiny transmitter, and a telemetering antenna. Most of these components are now available in countermeasures equipment. A system diagram appears in Fig. 9.

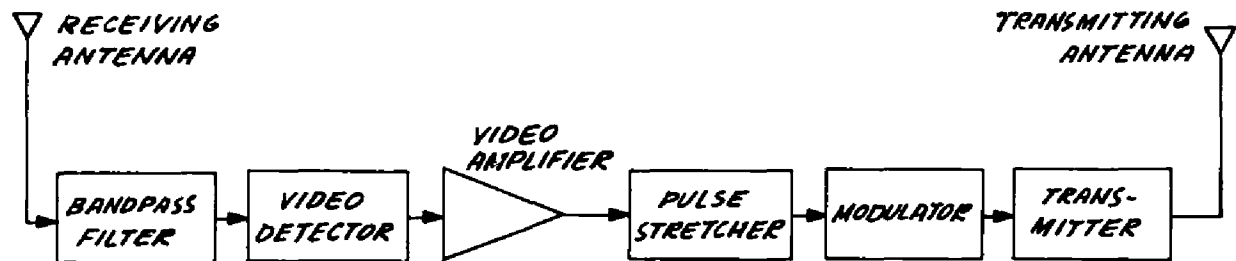


Fig. 9 - Satellite electronic reconnaissance system

Present crystal video systems have sensitivities several orders better than that required for this application. The satellite would have several microwave horns or spiral antennas oriented so that complete spherical coverage would be insured. These devices weigh but a few ounces each and it is felt that adequate coverage would be obtained by four antennas.

The bandpass filter limits the acceptance of the receiver to the frequency of the target. The bandwidth of this filter is designed to limit the data rate and, therefore, the required telemeter power by being narrow enough to eliminate essentially all signals except those particularly of interest. These devices are readily obtainable with the proper electrical characteristics at weights of but a few ounces.

The video detector and amplifier have been developed with transistor circuitry and the techniques of miniaturization. Present designs with bandwidths of 1 Mc are adequate to handle the expected signals. The power requirement for this device is approximately 100 milliwatts. The weight is about 4 ounces.

The pulse stretcher is a device that limits the information rate by compressing the bandwidth. It is useful in this system to conserve transmitter power by eliminating the redundancy of the target signal. The characteristics of this device are such that the

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original bandwidth of approximately 1 Mc is reduced to about 150 cps. These circuits have been successfully utilized for years to compress radar bandwidths to that of the human ear. Circuit-wise, the pulse stretcher consists of one transistor, one crystal diode, and an RC network. The power consumption is on the order of 1 milliwatt and the weight is about 1 ounce.

The modulator in the proposed system is a transistor circuit utilized to control the transmitter output in accordance with the intelligence derived by the receiver. The use of this device will limit the transmitter emission to those periods when the desired target signal is being received, and so conserve battery power. The complete modulator would weigh about 1 ounce and would require an average power of about 1 milliwatt.

The transmitter will send intelligence to the monitor stations when the desired target signal is being received by the satellite. It will consist of a piezoelectric oscillator for frequency stability, one frequency multiplier, and an output stage. The frequency of operation is chosen to be slightly below 100 Mc. This frequency will also assure line-of-sight propagation even from above the ionosphere at almost grazing incidence. This choice of frequency also allows high-efficiency transmitters utilizing transistors to be employed. Careful calculations involving fading show that for 90-percent reliability, a peak power of 100 milliwatts will be sufficient. Assuming a maximum 10-percent duty cycle and operation for only 25 percent of the time, i.e., only when near the Moscow area, and allowing for test interrogation from U. S. bases, an average power output of 2-1/2 milliwatts is required. The transmitter weight would be on the order of 6 ounces.

The transmitting antenna would be essentially the same as that designed for the Vanguard operation. The modification from 108 Mc to, say, 99 Mc would be simple.

The total power requirement for the proposed satellite is on the order of 100 milliwatts. For an assumed efficiency of 50 percent, on the basis of Vanguard information the battery would weigh about 4 pounds for an active life of 3 weeks. The possibility exists that with further development of the video amplifier a considerable saving may be effected.

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The total payload of the satellite will be on the order of 5 pounds. This is possible since the latest transistors and techniques are to be employed. Such a payload, accordingly, is compatible with a 40-pound satellite.

In conclusion, it is possible with well-known proved circuits and available components to build an Electronic Intelligence Reconnaissance Satellite capable of telemetering information from and about the Moscow defense complex to existing U. S. stations. The first device would operate on selected prime targets, would weigh less than 5 pounds, and would have an operational life of 3 weeks. This proposed system is feasible because of its simplicity, light weight, and availability.

It is estimated that a satellite of the type described above could be readied for flight testing within about a year, assuming that appropriate priorities would be assigned. Equipment for succeeding vehicles could be readied at the rate of about one per month.

The project outlined would require the highest possible security control and safeguards. This project has been discussed in detail with the cognizant personnel in the Office of the Director of Naval Intelligence. The requirements for such a program have been discussed and are being made the subject of official correspondence. This program would have one of the highest priorities assigned by the intelligence communities.

### 2.1.3.2.2 Electronic Intelligence Reconnaissance Satellite Systems (Secret)

The objective of this satellite would be to provide general coverage of the Soviet Union electronics installations.

In the early launchings the satellites would provide general coverage of a simple type essentially yielding only area deployment information. Later, more complicated satellites would be launched to provide specific information on selected intelligence targets.

A simple approach would be employed on the first satellites. The approach would become more and more complicated as the load-carrying capacity increased. When payloads of several hundred pounds become available, complete intercept systems incorporating complete signal analysis functions can be provided.

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In the initial stages of this project, receivers of the "wide-open" variety would be utilized which would allow for scanning the earth's surface with suitably disposed antennas in the missile to provide complete coverage regardless of the satellite's orientation. These receivers would feed simple audio recording systems which could play back the recorded signals at an accelerated rate when interrogated over the USA. All the techniques which would be utilized are available and within the state of the art at the present time.

In the interests of battery power the intercept receiver and recorder would not be operated except over potential enemy territory; the transmitter would only reply when interrogated again, running the recorder at accelerated rates only during the reply period. Later satellites would employ a multiplicity of functions and involve other systems of storage or memory.

An earth-encircling satellite would provide a means of electronic intercept over otherwise inaccessible territories of potential enemies. The satellite would be equipped with an intercept receiver and an electronic memory to permit recording of intercepted signals and subsequent retransmission of these signals by a radio transmitter aboard the satellite. The memory would be arranged to receive instructions through ground command transmissions. These instructions would activate the intercept function over enemy territory and the retransmission and command acceptance functions over friendly territory on a time basis, thus preventing the enemy from detecting the presence of the satellite and of interfering with its operation.

The intercept receiver would be arranged to cover radio frequency bands most likely to be employed by potential enemies. In order to minimize weight, different frequency bands could be covered on a sequential basis on successive passes of the satellite using a "wide-open" type of receiver for longer interval observations. A continuously scanning type of receiver might be employed if intermittent but more frequency-selective observations were desired. The memory would provide the necessary instructions for sequencing the various frequency bands of the "wide-open" receiver. These instructions could be altered subsequently through command transmissions to provide for any order of selection

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desired, or concentration on any specific frequency band. The selection could be arranged on a time cycle basis considered most effective.

The receiver would function to intercept signals and to receive instructions from a command transmitter alternately.

The selection of the "intercept" or the "interrogation" function of the receiver would be controlled on a timed basis from information existing in the memory. Prior to launching, initial timing instructions based on satellite launching performance would be placed in the memory. Subsequently, these instructions would be modified as indicated by information on the satellite's orbit obtained by tracking it. The timing would be so controlled that instructions could be received by the satellite only when it is within range of the command transmitter. Since the satellite would encircle the earth at fairly constant intervals changing very slowly it would be possible to adjust the timing quite accurately. A high radio frequency with line-of-sight transmission limitations would be chosen for the command transmissions. Thus a potential enemy would have no opportunity of observing the satellite transmissions while it was over territory under his control. Coding the information might provide further security where enemy agents might seek to obtain access to it through clandestine observations in friendly territory.

Through the tracking operation the position of the satellite over the earth at any particular moment will be known with fair accuracy. Even though intercepted signals will be somewhat modulated as a result of rotation of the satellite, considerable information relative to the locations of the sources of the signals should be obtained from variations of their amplitudes on a time basis.

A precedent and perhaps international policy has now been established for having a satellite pass over nations other than the launching nation. Subsequently planned military launchings will not be announced and hence will make satellites less conspicuous. Thus additional satellites for intelligence collection purposes will have good cover. The proposed Electronic Reconnaissance Satellite will be a passive device over potential enemy territory. It will emit tracking signals only when in range of strategically located tracking

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stations. Information transmission periods might be even more limited. Thus a potential enemy could not track the satellite by radio. It would be very difficult for the enemy to detect it visually without previous knowledge of its orbit. Radar detection would also not be very effective.

Explosive charges could be arranged to assure destruction of the satellite upon its return to earth either after performing its mission or in case of launching failure. A potential enemy thus could learn little about it through possible recovery of its parts.

The project outlined would require the highest possible security control and safeguards. This project has been discussed in detail with the cognizant personnel in the Office of the Director of Naval Intelligence. The requirements for such a program have been discussed and are being made the subject of official correspondence. This program would have one of the highest priorities assigned by the intelligence communities.

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### 2.1.4 COMMUNICATION SATELLITES

#### 2.1.4.1 Introduction (Secret)

A satellite might be employed as a repeater to provide a communication circuit over a distance of 2,000 or 3,000 miles.

Communications with submerged submarines are especially important. This is particularly true of atomic submarines which are capable of launching Polaris missiles carrying nuclear warheads. It is highly desirable that these submarines be capable of operating without the need for breaking the surface at all, even for the purpose of receiving orders to fire their Polaris missiles. Studies of this problem are now in progress.

#### 2.1.4.2 The Application of Earth Satellites to the Polaris Command Problem in the Arctic

##### 2.1.4.2.1 Introduction (Secret)

The application of satellites to the solution of the vital Polaris Command communication problem has been suggested. The following treatment examines some of the factors involved in designing and utilizing such a circuit.

Certain basic assumptions are taken, as follows:

(a) Practicable orbits by 1960 will have heights of 150-400 miles above the earth's surface.

(b) Polar orbits and highly inclined orbits are feasible.

(c) Satellite useful payloads will approximate 300 pounds.

(d) Polaris areas of initial prime interest are within the Barents Sea north of Scandinavia lying between  $70^{\circ}$  -  $80^{\circ}$  N latitude and  $15^{\circ}$  -  $60^{\circ}$  E longitude.

(e) Prime dependence is to be on communication from the continental U. S. That is, a system completely dependent for its operation on terminals within foreign and, at present, friendly countries, or on terminals located in U. S. territories, cannot be tolerated.

(f) Circuit delay must be minimized. Thirty minutes is an absolute maximum which must be reduced as the state of the art permits.

(g) Certainty of communication is of paramount importance.

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### 2.1.4.2.2 Choice of Frequency of Transmission (Secret)

The height at which the satellite travels requires that signals from it traverse most of the "effective" ionosphere. All radio waves are subjected to some alteration in passing into or through the ionosphere. These effects include changes in direction, intensity, splitting and polarization and produce signal distortions and variable coverage. The effects are frequency-dependent and their magnitudes are varied by natural phenomena chiefly associated with changes in solar radiation. In general, the effect of the ionosphere on the propagation of radio waves decreases with increased frequency and may be assumed small above about 80-100 Mc. Since ionospheric influences are not completely predictable, and since areas of acceptable circuit coverage may consequently expand, contract, or have gaps, it is considered that a frequency of operation of 100 Mc or above would prove most acceptable for this service. Antenna considerations on the ground make it desirable to choose a frequency near 100 Mc. Experience with Vanguard telemetering at 108 Mc will provide an invaluable guide to the suitability of the 100-Mc range for communication purposes.

Some suggestions for operation at very low frequencies have been made. These are based on the desire to communicate with completely submerged submarines. Frequencies suitable for this use lie below 150 kc. A brief and incomplete qualitative examination of the possibilities does not indicate a hopeful situation. Refraction and absorption effects may present very serious obstacles to the use of this frequency range from satellites. It is believed that sufficient data exists to calculate tentative quantitative information, and this should be done to settle the matter more conclusively. The provision of suitable satellite antennas in this frequency range would present some difficulty, but it probably can be accomplished at least in the upper part of the range.

### 2.1.4.2.3 Choice of Orbits (Secret)

By assumption (d) in section 2.1.4.2.1, the area of interest lies entirely within the Arctic Circle, hence highly inclined orbits (reaching about  $75^{\circ}$  N latitude) or polar orbits

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only will be useful. (Consideration of equatorial satellites having the same period of rotation as the daily rotation of the earth is eliminated by assumption (a).) At 300 miles altitude, the optical range is about 1600 miles, and at 400 miles it is 1800 miles. If useful ranges are assumed to be those where the satellite reaches at least 10 degrees above the horizon, these ranges are reduced to about 960 and 1150 miles, respectively. Thus, ignoring refraction, a useful range of about 1000 statute miles is assumed. The Scandinavian peninsula reaches to within about 1300 miles of the Pole, and other borders of the Arctic Ocean along the Eurasian continent are about equally distant. Hence, a strictly polar orbit cannot quite cover the entire Arctic at every passage. Observation of the Vanguard telemetering transmission can be used to check this tentative conclusion.

Use of the satellite as a repeater from which a message is transmitted as soon as received gives a total range of about 2000 miles. This range is inadequate to permit this mode of operation from the U. S. to the Polaris submarine, since the range required for this service approximates twice this distance. However, this mode might be useful for the less vital return circuit from the submarine to shore points or ships within reach. From the continental U. S., it appears that only the mode of operation in which the satellite stores and retransmits the message is workable. By this method, the satellite would be "loaded" as it passes the U. S., and it then would retransmit the message continuously for a period long enough to assure its receipt in the polar area. Assuming a loading zone stretching from 1000 miles east of Boston to 1000 miles west of Seattle, a total span of 95 degrees of longitude is covered. Hence, if delays of several hours are to be avoided, a satellite must be available at every 90 degrees. If a polar satellite would cover the entire Arctic, two satellites would suffice, but delays of nearly 3 hours might be encountered. This would occur when only a southward-bound satellite is available to the loading zone and it has left the loading zone just before the command message is decided upon. An entire revolution is then required before the message can be loaded and almost another before it is retransmitted in the Arctic. This time can be cut almost in half by using counter-rotating satellites in each orbit, a total of four.

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The use of inclined orbits means that more than two pairs of satellites must be used to insure coverage of the Arctic. Three pairs of 120 degrees will barely suffice and would permit maximum delays of about 1 hour and 40 minutes. With the conditions stated, these paths will pass less than 4 degrees from the Pole.

It is obvious from the foregoing discussion that the question of permissible orbits for solution of the communication problem posed includes also questions on maximum reliable range of transmission for which definite answers cannot now be offered. Therefore, it is important that these data be obtained during the Vanguard tests.

### 2.1.4.2.4 Transmitter Power (Secret)

The required transmitter power is dependent on several factors, such as practicable antenna gains, path attenuation, receiver sensitivity, ambient noise, and bandwidth. The following tabulation assumes a power output of 1 watt, a bandwidth of 10 kc, a receiver noise factor of 5 db, omnidirectional antennas, thermal noise only, and free-space path attenuation. Three frequencies are included for purposes of comparison.

|   | <u>20 Mc</u> | <u>100 Mc</u> | <u>400 Mc</u> |
|---|--------------|---------------|---------------|
| Transmitter output (1 w)                  | 0 dbw        | 0 dbw         | 0 dbw         |
| Gain of antennas                          | 0            | 0             | 0             |
| Path loss (2000 miles)                    | 129 db       | 142.8 db      | 154.8 db      |
| (1000 miles)                              | 123 db       | 136.8 db      | 148.8 db      |
| (500 miles)                               | 117 db       | 130.8 db      | 142.8 db      |
| Ktb noise at 10-kc bandwidth              | -163.5 dbw   | -163.5 dbw    | -163.5 dbw    |
| Receiver N. F.                            | 5.0 db       | 5.0 db        | 5.0 db        |
| Minimum Detectable Signal<br>(C/N = 0 db) | -158.5 dbw   | -158.5 dbw    | -158.5 dbw    |
| Carrier/Noise (2000 miles)                | 29.5 db      | 15.7          | 3.7           |
| (1000 miles)                              | 35.5         | 21.7          | 9.7           |
| (500 miles)                               | 41.5         | 27.7          | 15.7          |

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These data indicate that 1 watt, depending on the modulation chosen and types of fading, absorption, and noise encountered, would probably be satisfactory to 1000 miles at 100 Mc and useable at greater ranges with more frequent message error. Some decrease in bandwidth may be possible, depending on message complexity and length as well as the minimum time the satellite is above the horizon and receivable. Total doppler shift caused by satellite motion is a maximum of about 5400 cps at 100 Mc, and the bandwidth chosen must accommodate this plus the modulation, or provision must be made for doppler correction in the submarine, an undesirable complication if it can be avoided. The doppler at 400 Mc is about 22 kc maximum, hence the tabulated data given above for 400 Mc would require either partial compensation for doppler or correction by at least 3.5 db.

### 2.1.4.2.5 Security (Secret)

Assumption (g) of section 2.1.4.2.1 states that certainty of communication is paramount. An important element affecting this is the enemy's ability and opportunity to undertake jamming and spoofing. Conventional jamming may take the form of interfering with submarine reception by flying jammer transmitters over the Arctic areas known to be of interest. These might be borne by planes or high-altitude balloons. The only basic defenses against this type of interference are to change frequency – considered impracticable with present satellite limitations – to increase the transmitter power, or to use directional antennas aboard the submarine. This latter course appears to be the only feasible one.

Another type of jamming might be to jam the satellite receiver at the time of loading. This must be done from ships, bases, or planes within range of the satellite. Unfortunately, in assuming a loading zone reaching 1000 miles east from Boston, a similar jamming zone exists an additional 1000 miles to the east for the same satellite at the same time. This area includes some portions of Europe and European coastal waters. Again, the only real defense lies in increased transmitter power, this time of the shore-based terminal. Directional transmitting antennas are indicated.

Spoofing, or the insertion of false or altered messages by the enemy, can take two forms in the system outlined. He can load the satellite before it reaches the U. S. loading zone, making it impossible to enter a correct message when the loading zone is reached. If only northward-traveling satellites are utilized by the U. S., the false loading would have to be accomplished in the Western Hemisphere, south of the U. S. The enemy may also load other northbound satellites passing over his own territory to cause confusion. The defense against this type of attack would appear to lie in providing the satellite receiver with enough intelligence to tell a false from a true message. No security system of this sort exists, as far as is known, but some thought has been given to the same problem for another project, and it is believed that a method can be devised based on storing "unlocking" codes in the satellite which must be received correctly before a message is accepted for retransmission.

#### 2.1.4.2.6 Receivers (Secret)

It does not appear that the design of a satellite receiver would be difficult. The requirements for sensitivity are moderate since susceptibility to jamming will increase with sensitivity and shore-based loading transmitters can be powerful. As additional complications can be accepted at these terminals, correction for doppler can be introduced to narrow the required bandwidth and further reduce jamming effectiveness. In this case, a return circuit from the submarine to the USA would probably require a second receiver in the satellite. The Faraday rotation of polarization caused by passage of the wave through the ionosphere in the presence of the earth's magnetic field is of considerable magnitude at 100 Mc. Provision for accepting such signals must be included in antenna and receiver designs. This problem does not differ from that already faced by Vanguard.

#### 2.1.4.2.7 Power Supplies (Secret)

Vanguard experience will provide a guide to the direction of power-supply development. At present, silver-cell batteries are obtainable that produce 30 watt-hours per pound of weight, and claims as high as 90 watt-hours per pound have been published. Solar cells plus batteries are, of course, attractive, but their effectiveness and life in the satellite

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environment are unknown and unproved. Since the receivers of the satellite must operate continuously, a low-drain design is essential if a reasonable life is to be obtained. The coding elements must similarly be activated at all times. Storage and transmitter units are intermittent in operation and less critical. The necessity for low drain is apparent if one considers that 250 pounds of 30 watt-hours-per-pound batteries would last just a month with 10 watts continuous drain. Although no accurate prediction of life can be made at present, it is believed that several months operation could be attained with a conventional battery power source.

### 2.1.4.2.8 Conclusion (Secret)

The use of satellites to relay information and orders to submarines in the Arctic is considered feasible. Further experimental information is required before the minimum number of satellites can be established. For full coverage of the Arctic, maximum circuit delays will be about 100 minutes, and with certain assumptions as to range of transmission, six satellites in almost polar orbits would be required. Favorable experimental evidence would reduce this to four, but unfavorable results would require an increase. Reduction of maximum circuit delays to appreciably less than 100 minutes would also require a substantial increase in the number. A mode of operation can be devised to minimize enemy interference with the circuit. No insurmountable technical difficulties are foreseen, providing satellites of adequate size can be placed in the proper orbits. Further study of the mode of operation, orbits, potentiality of lower frequencies and other factors are considered necessary, and experimental evidence obtained from Vanguard operation must be obtained before final predictions as to reliability, security, and cost can be made.

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### 2.2 SUPPORTING RESEARCH

#### 2.2.1 INFRARED RECONNAISSANCE STUDIES (SECRET)

It seems probable that an infrared detector, or a set of them, sensitive in the wavelength band 0.5 to 3 microns, would detect the flame of a large rocket at a distance of several hundred miles at night. Daylight detection would be less certain of success because of the strong background of reflected and scattered sunlight, which also would limit detection of the flame by visible light devices.

The spectral region from 0.5 to 3 microns is tentatively chosen because a large fraction of flame radiation lies within this band and because photoconductive PbS cells of adequately high sensitivity and of short time constant ( $\sim 2 \times 10^{-4}$  sec) are readily available. Furthermore, there is a wealth of industrial engineering experience in the construction of systems utilizing these cells, e.g., the SIDEWINDER system.

##### 2.2.1.1 Radiant Intensity of Rocket Engines (Secret)

The first problem is to estimate the available flux of radiation reaching the satellite from the rocket. There are few data; only recently have intensive measurement programs been initiated on missiles in flight. It is likely, therefore, that not all available data are reported here, and it is also likely that the amount of information available will increase rapidly in the near future.

Butler and Harvey (2) of NRL measured radiation from a V-2 rocket flame in 1946, using a captured German PbS cell. This rocket burned alcohol with liquid oxygen; the thrust was 55,000 pounds. The signal was strong and fairly constant in intensity from launch point 10 miles from the detector until burnout at a rocket height of 19 miles and a slant range to the detector of 21 miles. The flux of radiation to which the PbS cell responded was  $4 \mu\text{W}/\text{cm}^2$ .

Curcio and Butler (3) measured the temperature of a 220-pound-thrust acid-aniline rocket flame from visible-light spectra and found that the flame radiates in the visible like a grey body at about  $2760^\circ\text{K}$  with a low emissivity, about 0.006.

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Measurements of total radiation in the band from 0.4 to 15 microns were compatible with values computed for a true temperature  $T = 2760^{\circ}\text{K}$  assuming constant emissivity  $E = 0.006$  throughout the infrared spectrum.

Curcio and Sanderson (4) extended these measurements on 220-pound and 440-pound-thrust engines, obtaining  $T = 2712^{\circ}\text{K}$  and  $E = 0.0055$ , in the visible spectrum, in general agreement with the earlier measurements. They computed a near-infrared spectrum from these values and found it in fair agreement with a spectrum measured by Wolfe (5). They used their own and Wolfe's results to estimate that, as an example of several such estimates, a lead sulfide detection system with 12-inch-diameter mirror should detect a 1,000-pound-thrust acid-aniline flame through a horizontal range of 91 miles at altitude 20,000 feet, with a signal-to-noise ratio of 9.

More recently a series of tests on the detection of Corporal acid-aniline flames (understood to have a 26,000-pound thrust) was conducted under the sponsorship of the White Sands Signal Corps Agency. This was called Project Terrycloth (6). Four lead sulfide devices with optical systems 8 to 12 inches in diameter were used at a distance of 120 miles. Eight rounds were fired, all at night. Strong signals were obtained in most cases. The instrument sensitivity was not stated, and quantitative data cannot be derived from the results.

In a few cases certain instruments were located 220 miles away, and in some cases these also detected the rocket flame. In other cases, the rocket probably did not cross the field of view of the instrument.

In the same operation R. M. Talley (7) of the Naval Ordnance Laboratory measured the infrared emission spectra of the flames, using a novel rapid-scan spectrometer. He found that more than 75 percent of the radiation lies in the region of the spectrum detectable with lead sulfide cells, i.e., in the 0.7- to 2.7-micron band. The values of spectral flux density at a horizontal distance of 6.2 km (3.85 miles) from the launch point are shown in Fig. 10 together with a relative response curve of PbS. The curve shown applied

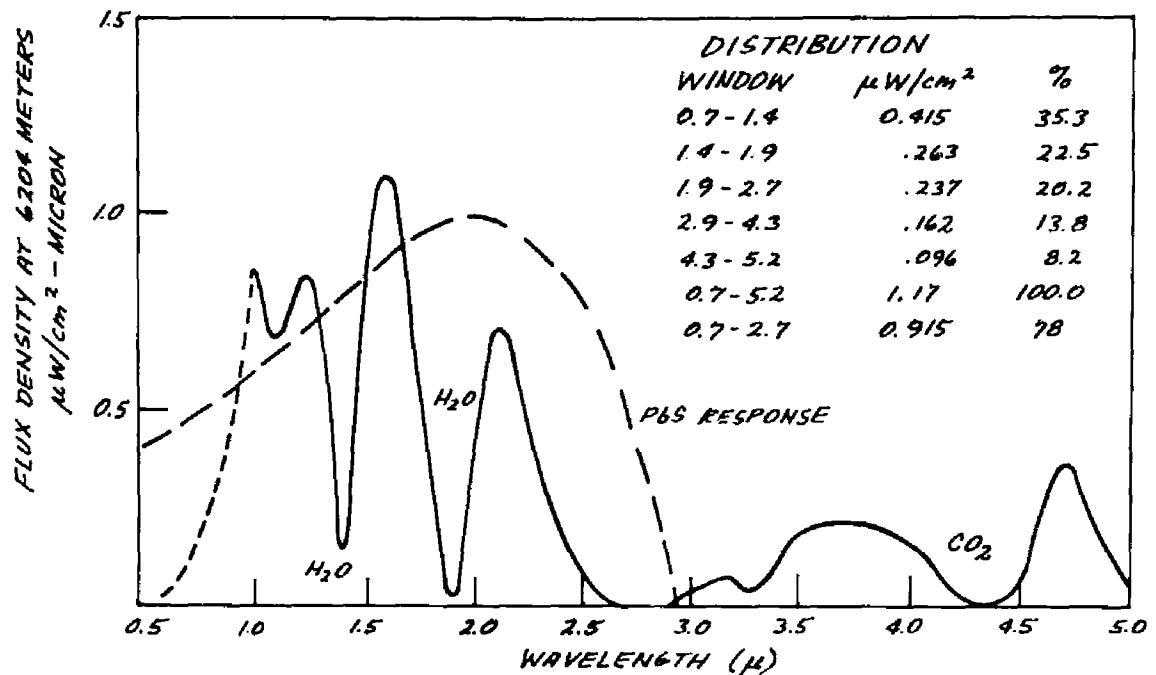


Fig. 10 - Spectral flux density from corporal flame at 3.85 milca

remarkably well both to plus 6 seconds and plus 22 seconds after ignition, and the report states that the spectral intensity remained essentially constant up to burnout at 54 seconds. Trajectory data and slant ranges were not stated; but the observation is in qualitative agreement with the measurements of Butler and Harvey that the V-2 signal remained constant until burnout at 19 miles altitude. It would be expected that the signal would decrease with the inverse square of the distance, and Butler and Harvey suggested that the constancy of signal as the rocket rose and the slant path increased may have resulted somehow from reduced attenuation by water vapor in the slant path.

Yates and Taylor (8) have measured the infrared emission spectra of jet aircraft with thrusts in the region from 6250 to 7800 pounds (F9F-8 and FJ3, respectively), and they have also used PbS radiometers filtered to respond between 2.0 and 2.8 microns and lead telluride (PbTe) radiometers filtered to respond between 3.7 and 5.2 microns, to measure signals from aircraft in flight. They found the signals 10 times stronger in the

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PbTe radiometer, owing to the strong 4.4-micron emission band of CO<sub>2</sub> in these engines. This would be the case also for kerosene, gasoline, or alcohol rocket engines viewed at high altitudes. However, PbTe cells require cooling with liquid nitrogen. Photoconductive cells sensitive to longer wavelengths are not yet adequately developed or available. The PbS cell is therefore considered to be the most practical detector for immediate use.

Some of the results listed above can be scaled up to provide inexact estimates of the flux of radiation from a missile flame burning, say, 5 or 10 miles above the ground. Attenuation of the signal by water vapor and carbon dioxide may then be ignored within the limits of accuracy of the computation. If, furthermore, the radiant output is assumed to be proportional to the engine thrust, all engines measured can be scaled up to an arbitrary size, say 200,000-pound thrust. Then the flux density of radiation to which a lead sulfide cell will respond at distance of 300 miles can be computed; the results are shown in Table 6.

TABLE 6  
Computed Flux Density at 300 Miles When Engine  
is Scaled up to 200,000-Pound Thrust

| <u>Vehicle</u>                   | <u>Flux Density<br/>(watts/cm<sup>2</sup>)</u> |
|----------------------------------|--|
| V-2 (alcohol-oxygen) (Ref. 2)    | $4.9 \times 10^{-8}$                           |
| Acid-aniline (Ref. 4)            | $6 \times 10^{-9}$                             |
| Corporal (acid aniline) (Ref. 7) | $1 \times 10^{-9}$                             |

The main value of these estimates is that they lie within the range of flux densities of radiation detectable with existing lead sulfide systems. The warning must be given that estimates such as these almost invariably prove more optimistic than the results achieved in practice. However, they may be applied to a proven PbS system, the SIDEWINDER homing head. Some of its characteristics have been described by Biberman, Chapman, and Schade (9). The system utilizes an optical mirror of focal length 8.9 cm and effective aperture f:1.13. Bench tests for acceptance of the units requires a signal-to-noise ratio of 3 to 1 and cell signal of 200 microvolts peak-to-peak for a radiant flux density of  $4 \times 10^{-9}$  watts/cm<sup>2</sup>. This input signal and voltage would provide several volts output, using

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the SIDEWINDER amplifier. Therefore, it appears probable that a system no larger or more sensitive than the SIDEWINDER system would detect a 200,000-pound-thrust engine at 300 miles, at night.

It may be remarked that while the SIDEWINDER system is neither the biggest nor the most sensitive PbS system that has been built, it is surely about the most successful. There does not seem to be justification for extrapolating expectations too far in terms of "equipment refinement." The SIDEWINDER system is reasonably refined.

It is possible that NOTS, Inyokern, already has tested SIDEWINDER performance against big missiles and that estimates of detection capabilities superior in accuracy to those given in this paper may already be available. Time has not permitted an investigation of this point.

This paper has not considered the intricate problems of orientation information, information storage and playback, etc. The following scanning considerations are cited however since they are of interest here.

### 2.2.1.2 Scanning Considerations (Secret)

In Fig. 11, if  $R = 3950$  miles and  $h = 300$  miles, then the slant distance to the horizon  $S$ , is 1570 miles,  $\theta_{max}$  is  $68^\circ$ , and  $\phi$  is  $22^\circ$ . The length of arc,  $ab$ , from horizon to horizon is  $2R\phi = 3350$  miles.

A satellite with a rotational period equal to its period of revolution and with its spin axis perpendicular to the plane of its orbit would always face the earth. At a speed of 25,500 fps in the orbit, a line of sight to earth would move at 4.5 mi/sec over the face of the earth. To allow, say, 1 millisecond on-target time for a point source, which would be adequate for a PbS cell, the minimum field of view in the direction of motion would subtend  $4.5 \times 10^{-3}$  miles = 24 feet. Since a rocket flame is very much longer than this (about 50 feet for the 40-foot Viking at take-off), a larger projected field of view would be required to allow for full exposure on rockets already tipped from the vertical when sighted. A field of view subtending 500 feet at the surface would be only 6.8 seconds of

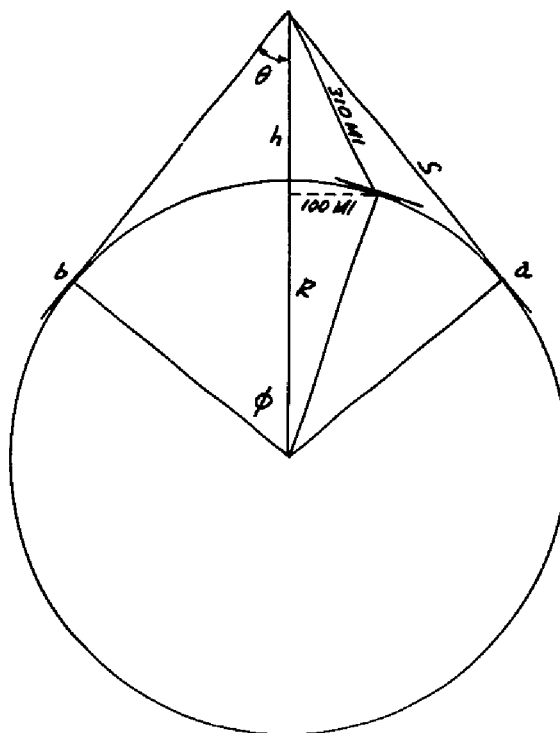


Fig. 11

arc in width. Practically, larger fields of view probably would be used. Location of the target rocket in the nadir of the satellite could be quite accurate with narrow field of view, provided that the field of view, traversing the surface of the earth at 4.5 mi/sec, crossed the launching point sometime during the burning period of the rocket engine. The chance of such a coincidence would be small. Widening the field of view to cover a longer arc on the earth in the direction of travel is feasible. A Schmidt or Maksutov optical system with a line of multiple PbS strips on its focal surface is indicated. Such systems enjoy the quality of good images over a field of view of about 40 degrees.

A schematic but inaccurate diagram of the system is shown in Fig. 12. If 40 individual PbS cells were mounted in the north-south line of motion of a polar satellite, each subtending one degree, the instantaneous field of view of the control cells would be 5 miles wide directly under the satellite; the terminal cells, 20 degrees to the north and south of the nadir, respectively, would view areas on the earth about 110 miles to the north and

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to the south. The array would thus cover 220 miles, more or less, in instantaneous view with 5 miles resolution in the nadir and poorer resolution toward the edges of the field because of the curvature of the earth.

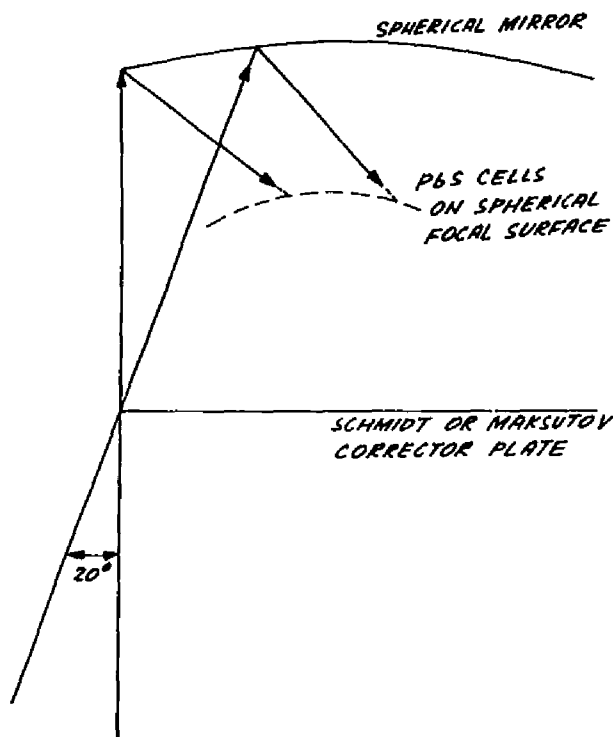


Fig. 12 - Optical system for rocket flame detection

A similar array of cells along the east-west line would sweep out a path of surveillance 220 miles wide at a rate of 4.5 mi/sec. A number of satellites would be required for constant surveillance, and only those viewing a rocket fired at night would be likely to identify a rocket launching.

A second scanning system could employ a normal rotational frequency of 3 revolutions per second, or 1080 degrees per second. Again, the favorable orientation of spin axis would be perpendicular to the plane of the orbit. In this case, the line of sight sweeps the earth in the nadir at 5400 mi/sec, and it sweeps from horizon to horizon (a to b, Fig. 11), in  $136^\circ / 1080^\circ = 0.126$  sec. Three detectors 120 degrees apart in the satellite would

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guarantee a continuous scan of the arc ab in the plane of the orbit. Two 40-degree fields of view at right angles would cover about 2000 miles in the north-south and in the east-west directions.

The further study of the use of satellites for infrared reconnaissance could well begin with an attempt to verify the basic capability. This might be done by means of a set of lead sulfide-cells installed in the satellite for viewing of our missile test ranges at AFMTC, Cape Canaveral, Florida; White Sands Proving Ground, New Mexico; and NAMTC, Pt. Mugu, California. Such an experiment could yield important test data. It might prove feasible to schedule the firing of certain of the rockets launched at these ranges in order to have them coincide with the satellite passage.

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### 2.2.2 OPTICAL DETECTION OF SUPERSONIC VEHICLES ABOVE THE ATMOSPHERE (UNCLASSIFIED)

Radiations in the far ultraviolet are expected from supersonic objects moving above the stratosphere. These radiations are produced by the primary atmospheric constituents, N and O. They result from two mechanisms, shock excitation and enhanced recombination of solar-ionized N and O. Both of these mechanisms can be expected to produce strong line emissions in the far ultraviolet region, e.g., NI 1243A, 1310A, 1311A, etc. and OI 1302A, 1305A, 1306A, etc.

Rocket surveys of atmospheric emissions encountered above 75 km indicate strong line and scattering emissions in the visible, near ultraviolet and far ultraviolet. Visible, near ultraviolet and infrared radiations have been anticipated from past ground observation of airglow and auroral spectra. The principal far-ultraviolet emission has been identified with solar system Lyman-alpha (1215.7A) radiation scattered by atmospheric atomic hydrogen. It is suggested that atmospheric ultraviolet emission excited by a missile could most easily be detected in the 1225A to 1350A band, which offers the lowest atmospheric background. The measured value of background flux is less than  $3 \times 10^{-8}$  ergs/(sec-cm<sup>2</sup>-steradian) at White Sands Proving Ground; it can be ascribed to fluorescence induced in the atmosphere by cosmic rays. The absence of the strong OI transitions yielding the 1300A lines is striking.

From these observations, significant advantages for the detection of ballistic guided missiles at these wavelengths at night are apparent.

1. There is low atmospheric background, less than  $3 \times 10^{-8}$  ergs/(cm<sup>2</sup>-sec-steradian).
2. Reliable high-sensitivity detectors are currently in use.
3. Jamming is difficult at these wavelengths.
4. Ground interference is eliminated by atmospheric absorption.
5. Object discrimination may be possible due to excitation mechanism.

An examination of the details of the excitation mechanism strongly indicate that the surveillance platform must have an altitude in excess of 300 miles. Furthermore, estimates of the height of the observed missile may be made by the characteristics of the

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observed signal. Below 80 km, sources are rapidly attenuated by atmospheric absorption. Between 80 and 130 km, shock excitation at these wavelengths is produced. This phenomenon has been detected during Aerobee rocket flights. Above 130 km, enhanced recombination of atmospheric  $N^+$  and  $O^+$  and other local surface effects become the dominant mechanisms.

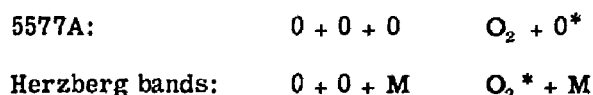
The promising possibilities offered by ultraviolet detection strongly indicate the need for continued and expanded rocket and satellite experimentation in this field. The basic feasibility of a ballistic missile detection system relying upon this phenomenon can be determined by a flight test as soon as a test vehicle can be made available.

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### 2.2.3 NIGHT CLOUD COVER, THE NIGHT AIRGLOW, AND THE AURORA (UNCLASSIFIED)

A satellite can easily be equipped to provide a worldwide survey of certain emissions in the night airglow. The results would be of great interest to meteorologists and upper atmosphere physicists. In addition, there is the possibility of using the night airglow around the horizon as a means for establishing the vertical, during darkness, and checking the vertical setting of a gyro.

The airglow is known usually to present rather large variations in time and space, and bright bands and patches sometimes travel across the sky. The changing sky pattern of airglow brightness has been studied from the ground by Roach, among others (10). The explanation for the variations is not obvious, but it appears to be connected with air movement and pressure changes in the upper atmosphere. From NRL rocket work, it is known that the 5577A line of OI in the green arises near 100 km (11). The Na D-lines arise near 85 km, and radiation from 2500-2800A from O<sub>2</sub> in the Herzberg bands exists near 100 km (12). The 5577A line and the Herzberg bands are believed to be produced by the following collision processes.



Both these reactions must be rather sensitive to pressure. The variations in the green line must somehow be related to airflow in the 100-km region, which may be sufficient to alter the above reaction.

Assuming that the pattern of emission is related to the circulation pattern at the particular level, it may then be possible to use the different airglow lines as tracers for the circulation at different altitudes. For example, the 5577A line should arise a little higher in the atmosphere than the Herzberg bands, since the third body in the former process is O and in the latter it may be either O<sub>2</sub> or N<sub>2</sub>.

In addition to the airglow, in the case of a polar orbit the photometers would measure the distribution of the polar aurora and provide a possibility of correlating the aurorae

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borealis and australis. This would be of great interest in connection with the process of excitation of the aurora.

The experiment would also give a measure of cloud cover at night, which would complement the data obtained by a day cloud-cover experiment, such as is planned in the Vanguard program.

It would be most important to attempt to correlate the results of the airglow and auroral experiments with observations made from the ground. During the IGY, there is set up a moderate-sized network of airglow monitoring stations, together with many stations for observing the aurora borealis (13). The entire airglow and aurora program would be unclassified, excepting probably observations made near the bright ring of airglow around the horizon, of the type that would bear upon the usefulness of the airglow for exact determination of the vertical.

In a 40-pound satellite, the experiment would consist of two photometers, each having a photomultiplier tube (RCA 1P21), an interference filter of approximately 20A bandwidth, and a lens, probably of 2 inches diameter. The outputs would be telemetered at as rapid a rate as the telemeter system would permit. One filter would pass the green line of atomic oxygen (OI) emitted in the night airglow near the 100-kilometer level, and also the green line in the case of a polar orbit; the other filter would pass a narrow band (20A) of the background light from stars and airglow near 5400A. Use of the 5400A photometer would make it possible to distinguish between an increase in signal due to clouds and a true increase in 5577A line emission, and to correct the 5577A photometer for airglow continuum radiation.

Transistorized power supplies for the photomultiplier dynodes are available, but electron-tube amplifiers requiring batteries for filaments will be necessary to reduce the high photomultiplier output impedance to the relatively low input impedance of the telemeter.

The two photometers would be mounted on the equator of the satellite, close together so as to view the same area simultaneously. If the satellite were pressurized, two

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windows of glass or quartz giving about a 2-inch-diameter viewing area would be required. Since the photometers would be very sensitive, a means would be provided to protect the photomultiplier tubes during the sunlit portion of the satellite's orbit. The volume occupied by the photometers would be approximately 3 x 6 x 6 inches, and their weight would be about 2-1/2 pounds. Power for the photometers would be obtained from mercury batteries that would occupy about 100 cubic inches and would weigh approximately 10 pounds.

In order to obtain the most data, which should be as complete and as easy to reduce as possible, the satellite should be launched so that its spin axis would lie in the orbital plane and be oriented parallel to a tangent to the earth at midnight, as in Fig. 13. This orbit restriction would still allow a reasonable tolerance on the firing time. The satellite itself should rotate about an axis in its orbital plane at from 4 to 10 revolutions per minute, in order to scan the sky at a reasonably slow rate.

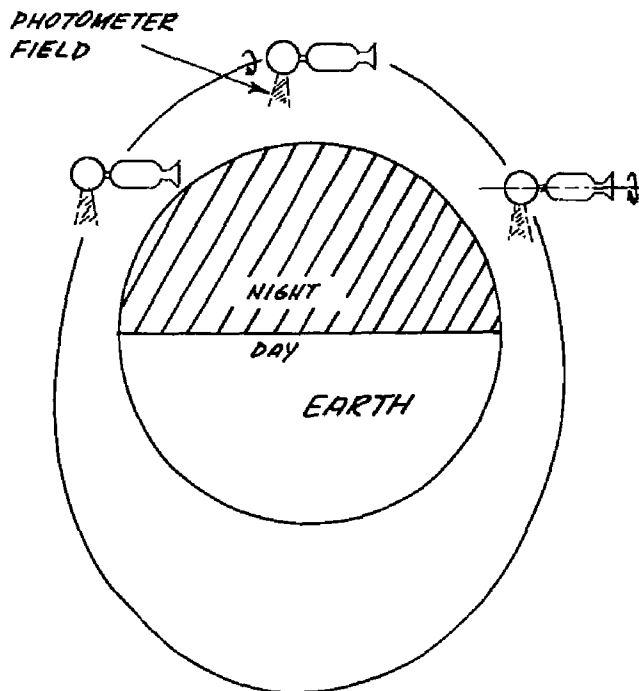


Fig. 13 - Orientation of satellite spin axis

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For a 300-pound satellite, it is assumed that more weight and space would be allocated to each experiment. The oxygen green line at 5577A would be monitored, along with the 5400A background region, as in the 40-pound satellite. In addition, the Na-D emission would be measured with a single photometer, and the ultraviolet in two wavelength bands between 3000 and 2000A. Another photometer would have a filter passing the red oxygen line at 6300A. And yet another, probably having a lead sulphide detector, would monitor the intense OH emission arising in a layer thought to be about 70 km above the earth's surface.

In the 40-pound satellite, the field of view of the photometers would be about 4 degrees. This would not provide sufficient resolution to determine the sharpness of the horizon from the point of view of indicating the vertical. Much narrower field photometers could be flown in the 300-pound satellite, and this would be the experiment more directly related to the problem of the vertical. The weight of the larger experiment might be 100 pounds.

The experiments would require about one year of preparation, with the full time services of 5 highly qualified scientists and 8 engineers.

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### **2.2.4 THE DETERMINATION OF THE VERTICAL (UNCLASSIFIED)**

Location of the vertical may be possible from a satellite by optical methods. The idea is to use the bright ring of light that will be presented by the horizon close above the earth's surface. The problem is to find a wavelength for which the ring is narrowest and least susceptible to local variations in height due to weather or terrain below. At night, some airglow radiation might be used. By day, the 2500-2600A region, arising from air scattering above the ozone layer, might prove to be satisfactory. There are many problems, and only after a good deal of research would it be possible to estimate the accuracy attainable with such a system. The basic research which must precede the development of such a system is included in the discussion of the preceding section.

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### 2.3 SCIENTIFIC RESEARCH

#### 2.3.1 GEOPHYSICS

##### 2.3.1.1 Magnetic Field Measurements from a Polar-Orbit Satellite (Unclassified)

The objectives of these measurements would be to obtain a complete map of the earth's main magnetic field, and the space distribution and length of auroral zone electric currents and disturbance currents in middle and low latitudes. The primary reason for pursuing these objectives are as follows.

The accuracy with which the magnitude of the magnetic field is known at all but selected locations on and above the earth, exclusive of local anomalies, is at best 1 percent. There are vast areas of the earth's surface where measurements of the field are extremely sparse. This 1-percent accuracy imposes a limitation on many practical uses of the earth's field. By means of "continuous" satellite measurements with an alkali vapor magnetometer, a complete mapping of the earth's field is possible. The accuracy of the map so obtained would be determined primarily by the accuracy of orbit locations vs time determinations. The error in field measurements would be less than  $\pm 1$  gamma ( $10^{-5}$  gauss). This accuracy would also lead to exact determinations of secular variations, if the experiment is repeated at intervals of several or more years.

Measurements obtained in high latitudes would permit continuous observations of the strength and distribution of auroral zone electric currents. This information would check and greatly expand the information obtained from the Fort Churchill rocket firings. Similarly, the measurements in middle and low latitudes would supplement the information to be obtained from the Project Vanguard magnetic field satellite.

The alkali vapor resonance magnetometer proposed by Varian Associates Inc. is ideally suited for these measurements. This magnetometer measures the absolute total scalar field independent of orientation, as does the proton precessional magnetometer. The principal advantage of the alkali vapor magnetometer in this application is that the power requirements are greatly reduced. A satellite alkali vapor magnetometer would probably

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weigh 1 pound and dissipate 1 watt of power. Reasonably small areas of solar cells would supply this power, and about 1 pound of batteries would be sufficient for power storage over the dark section of the orbit. Although intermittent measurement with data storage is feasible, it may be desirable to telemeter the signal continuously and make full use of recordings by radio amateurs. An alternative is to program the telemetering time and use increased power for reception at distant stations; this would be particularly advantageous if frequencies such as 20 Mc were used. As was shown by the first Soviet satellite, any one station would then receive about 20 percent of the measurements.

### 2.3.1.2 The Study of the Upper Ionosphere by Means of a Dual-Frequency Minitrack Satellite (Unclassified)

In the interest of obtaining additional ionospheric data, the use of a 40-Mc transmitter, in addition to the 108-Mc tracking transmitter, is proposed for one or more satellites. The general nature of the experiment is similar to that of the experiment described in Ref. 14. The use of the 40-Mc transmitter would provide data with 7.4 times the ionospheric effect of the data from the 108-Mc transmitter, and the 108-Mc transmitter would provide normal orbit determination. The five Minitrack stations now equipped to track on both 40 Mc and 108 Mc (Blossom Point, San Diego, Lima, Santiago, and Australia), would be able to pick up the satellite signal on 40 Mc at large zenith angles (where the effect of the ionosphere is most pronounced) prior to the time when it would enter the 108-Mc pattern. These antennas could then switch to the 108-Mc pattern, then back to the 40-Mc pattern for additional ionospheric data after the satellite passed out of the 108-Mc pattern. The 40-Mc antenna pattern is much wider in all directions than the 108-Mc pattern. Other tracking systems or ionospheric measurement stations could similarly obtain data on the 40-Mc signal that could be accurately correlated with the instantaneous position of the satellite for additional coverage in this experiment.

It is known that a local commercial laboratory can supply miniaturized 40-Mc transmitters suitable for satellite installation. Some modifications might be needed to enable

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them to meet the rigid Vanguard environmental specifications. The weight proposed is reasonable, however: less than 3 pounds for a life of several weeks. Development time is estimated at three months.

**2.3.1.3 A 12-Foot Satellite for Measuring Air Density and  
Providing a Readily Visible U.S. Object in Orbit (Unclassified)**

A 12-foot lightweight inflatable satellite would provide a splendid means for measuring air density and, at the same time, would present to the world a readily visible evidence of American satellite capability. Detailed studies of such a system have been made by the NACA. The results of some of these, presented below, were furnished through the courtesy of Dr. O'Sullivan of the NACA.

Estimates of the brightness of the balloon at various altitudes are as follows:

| <u>Altitude<br/>(Statute Miles)</u> | <u>Apparent Magnitude<br/>at Zenith</u> |
|-------------------------------------|---|
| 200                                 | 1.4                                     |
| 300                                 | 2.3                                     |
| 400                                 | 2.9                                     |
| 500                                 | 3.4                                     |
| 600                                 | 3.8                                     |
| 700                                 | 4.1                                     |
| 800                                 | 4.4                                     |

If the balloon is not at zenith, a correction should be added to the magnitude, as follows:

| <u>Angle from Zenith<br/>(degrees)</u> | <u>Magnitude<br/>Correction</u> |
|--|---------------------------------|
| 0                                      | 0                               |
| 20                                     | 0.06                            |
| 40                                     | 0.15                            |
| 60                                     | 0.27                            |
| 80                                     | 0.75                            |

For maximum rigidity for a given weight, the balloon skin would be made of a sandwich type construction employing 0.0004-inch aluminum, 0.001-inch Mylar, and 0.004-inch aluminum. The 12-foot sphere would weigh approximately 12 pounds. The compressed air

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bottle (equivalent in volume to a 6.5-inch-diameter sphere) would weigh about 2.9 pounds and have a storage pressure of 1500 psi.

The balloon would also be equipped with a Minitrack transmitter to insure acquisition and assist in the early stages of the tracking. The Minitrack items would weigh about 2 pounds. The entire satellite would thus weigh somewhat less than a regular Vanguard satellite. The balloon would be packed in a container which would fit the present Vanguard launching vehicle nose cone configuration. The package would be projected from the third stage rocket after burnout in a manner similar to that of the regular Vanguard satellites. The balloon would then be inflated. The balloon system could be gotten ready in about six months, assuming proper priorities.

### 2.3.1.4 A Satellite Experiment to Measure Charge Densities and Ionic Compositions in the Upper Atmosphere (Unclassified)

Information which can be obtained from this proposed experiment includes:

1. Ion densities (negative and positive),
2. Ion temperatures,
3. Electrical potential of the satellite,
4. Impact distribution of ions upon the satellite,
5. Relaxation time of the atmospheric ionization (day-to-night), and
6. Identity of the ions present.

The equipment necessary to carry out this experiment is shown in Fig. 14. The principle of operation of the device is as follows: The magnetic field of the bar magnet prevents electrons from the medium outside of the satellite from reaching the entrance grid E. Ions from the outside of the satellite pass through grid E. If they have enough kinetic energy due to momentum perpendicular to the plane of the grids to overcome the potential difference between grid C and grid E, the ions pass grid C and are collected by the collector A. The voltage on grid C is swept between ground (i.e., the voltage of both grid E and collector A), and about 20 volts positive, as shown in Fig. 15. The voltage on

grid B is swept in the reverse sense and through just enough range to neutralize the charging current induced on collector A due to the varying voltage on grid C. The voltage on grid D is swept in the same manner as the voltage on grid B, in order to neutralize the electric fields outside the satellite in front of grid E caused by the varying voltage on grid C.

Suppose positive ions pass grid E with the kinetic energy of relative motion, which for atomic oxygen ions is about 7 ev. A repelling potential on grid C will turn them back unless they have energies greater than  $eV_C$ , hence they will not be collected by collector A and registered by the electrometer tube. If the voltage on  $V_C$  is made to vary as shown in Fig. 15, the current at the collector will appear as shown in Fig. 16, if only the one type of ion were present, the temperature of the ions being related directly to the slope of  $I_A$  halfway down on the cutoff.

If more than one type of positive ion is present,  $I_A$  will vary as illustrated in Fig. 17, and each variety of ion may be identified, because the large component of velocity of each is the same and is known (i.e.,  $= 10^6$  cm/sec).

To indicate the order of magnitude of the current to be expected, assume an ion density of  $10^5/\text{cm}^3$  and an aperture of  $10 \text{ cm}^2$ . For a satellite velocity of  $10^6$  cm/sec,

$$(I_A)_{\text{max}} = 1.6 \times 10^{-7} \text{ amp.}$$

which is easily measurable. If negative ions are present, a negative  $I_A$  will persist after all positive ions have been blocked, hence positive  $I_A$  may be corrected accordingly.

It can be seen that if the satellite were charged negatively the cutoff potential  $V_C$  would be displaced to larger values because of the greater energies. In this way the satellite potential can be measured. It should be noticed that electrons have been eliminated from consideration by means of a small bar magnet  $M$  located behind the grids.

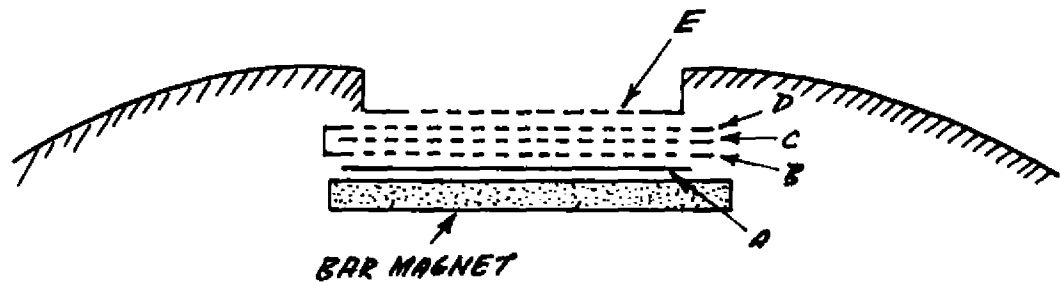


Fig. 14 - Satellite instrument for measuring upper atmospheric charge densities and ionic compositions

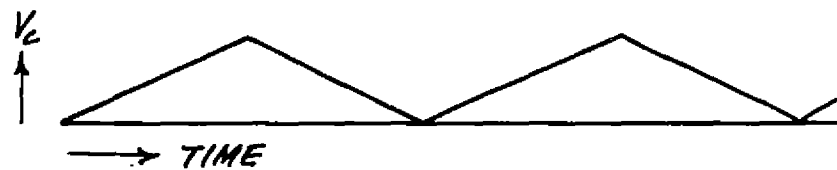


Fig. 15 - Voltage on grid B

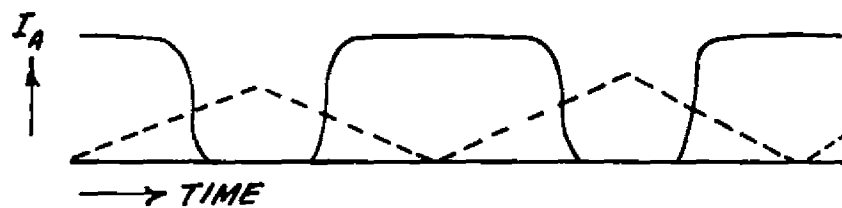


Fig. 16 - Current on collector A for a single type of positive ion

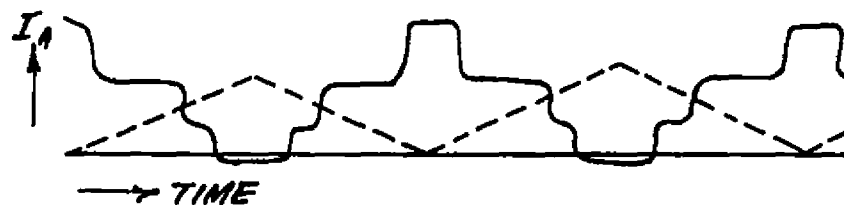


Fig. 17 - Current on collection A for more than one type of positive ion

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It should be stressed that this device does not produce any electric field outside of the satellite. In the operation of this device, the telemetering transmissions must be turned off while the memory in the satellite stores information, since it is almost certain that a transmitter will greatly distort the ion energy spectrum. It seems reasonable that this experiment can be performed in a 20-pound satellite following the polar orbit. The estimated time for launching preparation is about 9 months.

### 2.3.1.5 Certain Satellite Studies of the Earth's Atmosphere (Unclassified)

A great deal can be learned about the upper atmosphere of the earth from a satellite. The advantage of satellites over rockets is that they permit a continuing worldwide survey to be made of a particular phenomenon or quantity. This would be of inestimable value to climatology and meteorology.

The measurements may be divided into two categories: Those using the sun as a light source and studying its attenuation in various spectral regions during the process of sunset on the satellite; and measurements of the light emitted by the earth and its atmosphere, including reflected and scattered sunlight, and luminescence of the atmosphere.

#### 2.3.1.5.1 Measurements Using the Sun as a Light Source (Unclassified)

The following surveys might be made with the sun as a light source:

1. A survey of the vertical distribution of atomic oxygen in the earth's atmosphere, by measuring the extinction of the solar OI triplet at 1300A by the atmosphere during sunset.
2. A survey of the vertical distribution of O<sub>2</sub> in the region above 100 km, where dissociation gradually takes place, using photon counters in the region 1400-1500A and measuring the extinction during sunset.
3. A survey of the vertical distribution of ozone, at the top edge of the layer, by measuring the extinction of 2500A and 3000A during sunset.

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4. A survey of the vertical distribution of atomic nitrogen, by measuring the resonance lines of NI, near 1135A, which are probably present in sunlight.

5. A survey of the H<sub>2</sub>O content in the upper atmosphere, by measuring the 8-13 $\mu$  region, which is not absorbed, and the 5-7  $\mu$  region, which is strongly absorbed, during sunset. Another possible wavelength is the 1.9  $\mu$  water vapor band, which can be isolated with an interference filter.

6. A survey of the density distribution in the upper atmosphere by measurement of various x-ray regions during sunset.

### 2.3.1.5.2 Light from the Earth and its Atmosphere (Unclassified)

The following measurements might be made of the light emitted by the earth and its atmosphere

1. The night airglow. The limit has probably been reached in studying the spectrum of the airglow from the earth. More could be learned from a satellite because this platform combines long exposures with the chance of viewing the airglow edgewise. An order of magnitude in intensity could be gained and this would make it possible to obtain more highly resolved spectra. These, in turn, are exactly what are needed to identify the photochemical processes occurring in the upper atmosphere. One would also extend the work into the ultraviolet below 3000A, which cannot be studied from the earth. Rockets are not very useful for this purpose because very long exposures are required. One would either try to recover the spectrograph, or, scan photoelectrically with some sort of storage device, or develop the exposed plate in the satellite, as with a Land camera, run it through a simple densitometer, and telemeter the signal.

2. Worldwide survey of night airglow. The night airglow is well known to present changing, complicated patterns which must somehow be related to the meteorology of the 70 to 150 km region. From a satellite in a polar orbit the airglow distribution could be surveyed all over the world on a continuing basis. This could be done in the several important wavelengths: 5577A and 6300A from atomic oxygen, arising from 100 km and,

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probably, above 160 km respectively; and the 5893A Na-D line. The continuum: the infra-red radiation from OH, and selected regions in the ultraviolet. The study should include wavelength regions in the extreme ultraviolet also. All these radiations arise from different processes and from different levels in the upper atmosphere. A study of their worldwide pattern may reveal flow lines in the upper atmosphere and tie in with weather changes, and also the transport of atomic debris.

3. A survey of atmospheric density. Far ultraviolet luminescence measurements of the sunlit atmosphere, scanned from the edge of the earth toward zenith, may provide information on atmospheric density and scale heights up to 1000 km, and on the extent of atmospheric density fluctuations over the earth. It may be possible to measure the atmospheric density distribution by means of the Rayleigh scattered light. Wavelengths would be selected for which the earth's surface would appear black, so that the situation would not be complicated by light from below. Possibilities are: 2500-2600A, which is completely absorbed by ozone; 2100A, which penetrates only to about 10 km; and 1400-1500A, which is attenuated strongly by O<sub>2</sub>, reaching an altitude of about 100 km. It would be very interesting to see what the earth looked like in light of these wavelengths. It would be a thin bright ring and little else. The distribution across the ring would give the atmospheric density function.

4. A survey of the total ozone. This could be done by measuring the ratio of 3100 to 3300A light returned from clouds or snow, directly below. Possibly other terrain could be used also. These measurements are greatly sought by meteorologists, who have set up a small worldwide network for the IGY. From a satellite many times more data could be obtained, and in all probability a correlation with airflow patterns could be established.

5. Survey of the stratosphere for debris. The brightness of the sky in the neighborhood of the sun depends on the amount of material present, and is a sensitive indicator of large particles such as smoke and dust. An extreme example in the ring around the sun following great volcanic eruptions. The idea is to photometer the sky near the sun using

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a coronagraphic instrument such as was devised by Evans. As the sun approaches the horizon, before setting, its rays pass through the stratosphere. The sky should be very dark, but if it brightens more than is consistent with the air density, this would indicate the presence of debris.

6. Use of upper-atmospheric water vapor as a tracer for atmospheric circulation. It would be possible to measure the radiation from water vapor in a region of great opacity, such as the 5-7  $\mu$  band. The intensity would be a function of the temperature of the water vapor in a very high region of the atmosphere. A survey of this would give a map of the temperature of this "level" in the atmosphere. This might be a good way to study the streams of warm air that circulate from the equator to the Arctic and Antarctic.

7. Auroral luminescence. From a satellite in a polar orbit, measurements can be made of the Lyman-alpha and adjacent far ultraviolet, and x-ray luminescence associated with the production of the aurora borealis. Such measurements are essential to our understanding of the processes that trigger the aurora and sustain it.

### 2.3.1.6 Satellite Atmospheric and Particle Studies

The experiments listed below are not discussed in as much detail as are many of those discussed in the foregoing section. Nevertheless, they are generally based on proved rocket techniques. They are indicative of the many possibilities which can be exploited once adequate satellite vehicle capabilities are available.

#### 1. Atmospheric density and temperature experiment.

Purpose - to obtain density and temperature as functions of altitude, latitude, and time.

Method - use ionization gauge on spinning satellite to measure directly the density and temperature/mass ratio.

Total weight 40 pounds, total volume 0.5 cu.ft. This experiment can be ready in about 1 year.

2. Positive ion composition experiment.

Purpose - to determine positive ion composition,  
mass 1-35.

Method - use radio frequency mass spectrometer.

20 pounds, 0.5 cu ft.

This experiment could be ready in about 1-1/2 years.

3. Auroral particles experiment.

Purpose - to study time variation of particles which  
produce aurorae.

Method - scintillators and photomultipliers in polar orbit  
satellite. 20 pounds.

This experiment could be ready in about 1 year.

4. Oxygen and water vapor altitude distribution experiment.

Purpose - use eclipse of sun by earth's atmosphere to  
obtain height gradients of oxygen and water vapor in the  
earth's atmosphere.

Method - measure solar radiation at 5 mm or 2.5 mm near  
grazing incidence, in satellite orbit of at least 300 miles  
altitude. Weight, 150 pounds, plus power supply capable  
of 200 watts, exclusive of telemetering.

### 2.3.2 SOLAR PHYSICS (UNCLASSIFIED)

Many experiments involving the sun are not only important but feasible. Some of these are discussed briefly below.

The availability of earth satellites makes possible a great many different kinds of measurements of the radiation from the sun, and of the structure and meteorology of its atmosphere. The type of experiment depends on the size of the vehicle.

The advantage of a satellite is the long observing time, which makes it possible to study dynamic changes in the sun and to study solar radiations with more sensitivity and resolution than can be achieved with rockets.

#### 2.3.2.1 Variation of Solar Ultraviolet Below 3000A (Unclassified)

Although the solar energy below 3000A is relatively small, not a few meteorologists believe that it has an important trigger action on weather through heating of the upper atmosphere. It is possible to determine whether there are significant variations of solar output below 3000A, by means of a simple experiment for a satellite. The solar ultraviolet output from 3000A to approximately 2000A would be compared with the energy in a narrow band in the visible region. Thus one would have two photometers whose outputs would be adjusted to be nearly equal and which would be bucked. Only one telemeter channel would be required and this only part of the time.

The light-collecting optics would be identical, consisting of recessed integrating spheres, translucent balls, or perhaps just flat photocathode surfaces arranged so that the signal output from bucked detectors would be independent of aspect relative to the sun, thus avoiding the difficulties of aspect determinations. By use of one or more additional pairs of units, the signal could be made to hold up even when sunlight no longer fell on the first unit.

A recently developed  $\text{Rb}_2\text{Te}$  photocell or equivalent would be used for the ultraviolet detector, and a narrow-band optical filter with a common type of S-4 detector for the visible. It is expected that the equipment for this experiment, exclusive of telemetry but

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including its own power supply, would weigh approximately 10 pounds and thus be suited to a 40-pound satellite. It could be carried out in about a year or two with four well-qualified full-time scientists and engineers.

### 2.3.2.2 Solar Spectrum (Unclassified)

Techniques are available utilizing selective filters, monochromatic photon counters, ionization chambers, and secondary emission multiplier tubes for mapping out the entire range of solar radiation in the invisible ultraviolet and x-ray regions. From rocket measurements the distribution of radiation is fairly well known for a quiet sun. The major advantage of a satellite would be the possibility of continuous observation over a period of a year or longer of the range of fluctuations in emission in all parts of the spectrum. Such experiments can be designed in simple form to measure the most striking phenomena, such as solar flares, in minimum-weight satellites. In heavier satellites the degree of coverage can be proportionately increased and the dynamic range expanded. In heavier satellites it will be possible to fly spectrographs with photoelectronic detection of key emission lines in the solar spectrum. Such experiments are of fundamental interest in the astrophysics of the sun and have a direct relationship to many atmospheric processes such as the ionization of the various regions of the ionosphere, the photochemistry of the atmosphere, and the conversion of solar energy into atmospheric winds and turbulence.

In a moderate-sized satellite it would be possible to fly a high-dispersion spectrograph, kept directed at the sun with a pointing control. The spectrographic instrumentation could be developed rather easily. For example, a 1-meter grating spectrograph, used in conjunction with an appropriate photoelectronic detector, could be used to study the solar spectrum from 2000 to 100A. Two spectrographs might be required to cover this entire region, but a complete scan could probably be accomplished in 10 minutes or less. This would permit monitoring changes in the various line intensities. Fainter lines and many more lines could be detected in this way than in the limited time of a rocket flight. Relative intensity values would be obtained which would be more meaningful than

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those gotten from rocket flights, wherein there is usually some chance of absorption by atmosphere constituents.

### 2.3.2.3 Studies of the Solar Atmosphere (Unclassified)

Solar weather can be mapped continuously by satellite-borne equipment. Pictures of the disc of the sun, and the corona, could be obtained in many key wavelengths. For some purposes, such as Lyman-alpha measurements, a coronagraph type of instrument would be used, scanning the monochromatic solar image photoelectrically. For other radiations a spectroheliograph or monochromatic camera, with a photoelectrically scanned image, would be preferred.

The wavelengths would be chosen so as to study the dynamic changes taking place at different levels in the solar atmosphere, and in the neighborhood of spots, flares, surges, prominences, and so on. The result would be a three-dimensional picture of the changes occurring in the sun's atmosphere.

The spectral lines of particular interest are:

Lyman-alpha, 1216A; beta, 1026A; and the Lyman discontinuity, 910A.

He II, 304A; 256A, and the discontinuity.

He I, 584A; He II, 1640A corresponding to H.

Mg X, 625A; various lines of highly excited Ne, O, N.

Soft and hard x-rays.

### 2.3.2.4 Solar Variations (Unclassified)

For many years the question of whether the sun is a variable star or not has been investigated. Of course, were it to vary significantly, our weather and climate should respond. It has been shown rather conclusively, by a restudy of the Smithsonian measurements, and by new measurements on Uranus made at Flagstaff, that variations in the total output of the sun must be less than one percent.

A 300-pound satellite would make possible the monitoring of the output of the sun in several wavelength regions as follows:

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1. Total output, i.e., the solar constant: An accuracy of 0.1 percent should be attainable from a satellite, because of the absence of attenuation by the atmosphere.

2. The total ultraviolet below 3000A: This could be done with a photocell and proper filter. Although the energy involved is small some meteorologists believe that it has an important trigger action on weather, through heating of the upper atmosphere.

3. The total ultraviolet below 1750A: These wavelengths disassociate oxygen, giving rise to ozone and effecting much of the chemistry of the upper atmosphere. Variations, if present, may have a critical connection with the behaviour of the ozone layer and the atmosphere above 80 km.

### 2.3.2.5 Magnitude of the Sun (Unclassified)

The magnitude of the sun is a basic astronomical physical quality that is not known to better than about 0.1 mag.

It would be possible to determine this quantity more accurately than has hitherto been possible by launching a good-sized spherical satellite whose reflectance is known. This would act as an attenuator. The satellite would be, perhaps, 2 magnitude and could be compared exactly with stars of known magnitudes by standard astronomical methods.

### 2.3.2.6 Satellite Radio Astronomy Studies (Unclassified)

The experiments listed below are not discussed in as much detail as are many of those discussed in the foregoing sections. Nevertheless, they are generally based on sound principles of radio astronomy. They are examples of the many possibilities which can be exploited once adequate satellite vehicle capabilities are available.

#### 1. Solar corona composition

Purpose - to determine the neutral and positive ion composition of the solar corona, mass 1-35.

Method - radio frequency mass spectrometer. 40 pounds, 0.5 cu. ft.

This experiment could be ready in about 1-1/2 years.

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**2. Solar brightness distribution**

**Purpose -** measure solar brightness distribution and brightness temperature at 0.5 mm.

**Method -** one-ft antenna with infrared detector at focus would give 7 min of arc resolution. Weight 50 pounds; power, 50 watts; both exclusive of telemetering.

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### 2.3.3 INTERPLANETARY AND COSMIC STUDIES

#### 2.3.3.1 A Satellite Experiment to Determine the Distribution of Hydrogen in Space (Unclassified)

It is proposed that the intensity of hydrogen Lyman-alpha radiation (1215.7A) received directly from the sun and the resonance radiation of the same wavelength produced by sun-lit hydrogen atoms in space be measured simultaneously. A relatively insensitive photon counter would be used to monitor the intense 1216A radiation emitted directly by the sun. This radiation pours out through space and is scattered in all directions by hydrogen atoms. At the same time, hydrogen ions are produced from atoms in space by the shorter solar ultraviolet and x-ray radiation or are ejected into space directly from the sun. A fraction of these protons and electrons will recombine into excited states, leading to a comparatively slowly varying background of the same Lyman-alpha wavelength. An extremely sensitive photon counter can be used to measure the sum of scattered and recombination radiation. When a large flare appears on the sun, the scattered radiation will promptly increase in direct proportion to the growth of Lyman-alpha in the flare. The effect of recombination radiation on the intensity will be much slower, so the contribution due to scattering by neutral hydrogen in space can be distinguished from the recombination radiation.

Any theoretical computation of the radiation intensity requires, among other things, a knowledge of the absorption cross-section for resonant scattering, the ionization cross-section, the recombination coefficient for radiative capture of electrons by protons, the solar intensity, and, of course, the hydrogen atom and ion densities. Some rough estimates have been made using the following numbers:

Solar flux density of Lyman-alpha at earth =  $1 \text{ erg cm}^{-2} \text{ sec}^{-1}$ .

Scattering cross-section in core of Lyman-alpha line =  $5 \times 10^{-14} \text{ cm}^2$  .

Recombination coefficient =  $1.5 \times 10^{-12} \text{ cm}^3 \text{ sec}^{-1}$ .

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Examples of some results derived from these figures are:

1. The scattered intensity in quanta  $\text{cm}^{-2} \text{ sec}^{-1}$  from a  $2\pi$  hemisphere looking away from the sun is about  $10^{10} N_H$  where  $N_H$  is the neutral hydrogen density, assumed to be uniform in space.

2. The intensity of recombination radiation varies from  $2N_H + 2$  to  $7N_H + 2$  quanta  $\text{cm}^{-2} \text{ sec}^{-1}$ , looking away from the sun, depending on whether the ion distribution near the earth is inverse square or inverse first power with distance from the sun.

Without knowledge of the ratio of neutral to atomic hydrogen, a single measurement of Lyman-alpha intensity from space will not make it possible to estimate the hydrogen distribution. The correlation with flare activity, however, permits a simple separation of resonant and recombination radiations.

The experimental measurements in the satellite would be performed by photon counters (15). Such tubes operate according to the familiar Geiger counter principle, except that discharges or counts are triggered by ultraviolet photons which photoionize the gas contents. A novel feature of the tubes developed by the Naval Research Laboratory is the elimination of the long-wavelength response ordinarily obtained from the cathode surface. This is achieved by using an electro-negative constituent in the gas mixture. Slow electrons photoelectrically ejected from the cathode are quickly attached to form negative ions, which drift to the anode without triggering counts. The desired short-wavelength threshold is obtained by including a gas component which has its photoionization threshold in the far ultraviolet. Nitric oxide serves both functions in a very efficient detector for Lyman-alpha. Its photo-threshold is at 1340A, and the ionization cross-section at the Lyman-alpha wavelength is very high. In combination with a lithium fluoride window, it is sensitive to the spectral region from 1100A to 1340A, and it is possible to obtain quantum yields of a few percent at the Lyman-alpha wavelength with comparatively negligible response to longer-wavelength solar radiation.

The insensitive photon counter is prepared with chlorine as the negative-ion former. Although chlorine itself is not photoionized by Lyman-alpha radiation, it forms trace

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compounds which are ionizable. Instead of obtaining yields of  $10^{-2}$ , as is the case with NO, the halogen-quenched tubes have yields of about  $10^{-7}$  between 1100A and 1300A. This low yield is adequate to produce several hundred counts per second when the tube views the solar Lyman-alpha radiation directly.

To return to the figure quoted above of  $10^{10} N_H$  quanta  $\text{cm}^{-2} \text{sec}^{-1}$  from  $2\pi$  steradians, a sensitive counter with a yield of 1 percent and a  $1\text{-cm}^2$  window would produce  $10^8 N_H$  counts  $\text{sec}^{-1}$ . If the field of view is restricted to one-thousandth of the hemisphere, the counting rate will be  $10^5 N_H$ . A concentration  $N_H$  of  $10^{-4} \text{cm}^{-3}$  would give a measurable intensity, and in the event of a flare the intensity would rise proportionately. The recombination radiation should then be observable, if the concentration is about 10 ions  $\text{cm}^{-3}$  or greater for almost any distribution extending over the dimensions of the solar system. Restriction of the field of view would permit distributed sources of hydrogen in local space to be mapped. It should also detect celestial sources of Lyman-alpha radiation in the background. One nighttime rocket flight has already been carried out.\* This flight showed that high counting rates of Lyman-alpha from space can be obtained with a 5 degree field of view.

A block diagram of the circuitry proposed for the satellite is shown in Fig. 18. This system would operate only during telemetering. At the present stage of development this circuitry, including detectors and batteries to provide five hours of operating life, would weigh less than 900 gm. If the spin rate is to be as high as one per second, the frequency response of the individual channels should be no less than 200 cps. The rate meters indicated in Fig. 18 have a useful dynamic range of over three decades. This is accomplished by counting individual events at low rates and using circuits and counter dead times to give a compressed scale at the highest rates.

A lightweight optical aspect system compatible with single-channel telemetering and low power drain has been developed and used successfully in several rocket flights. The essential features of this system are a highly collimated high-sensitivity photocell

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\* NRL Aerobee 25, 2:00 A.M., November 17, 1955

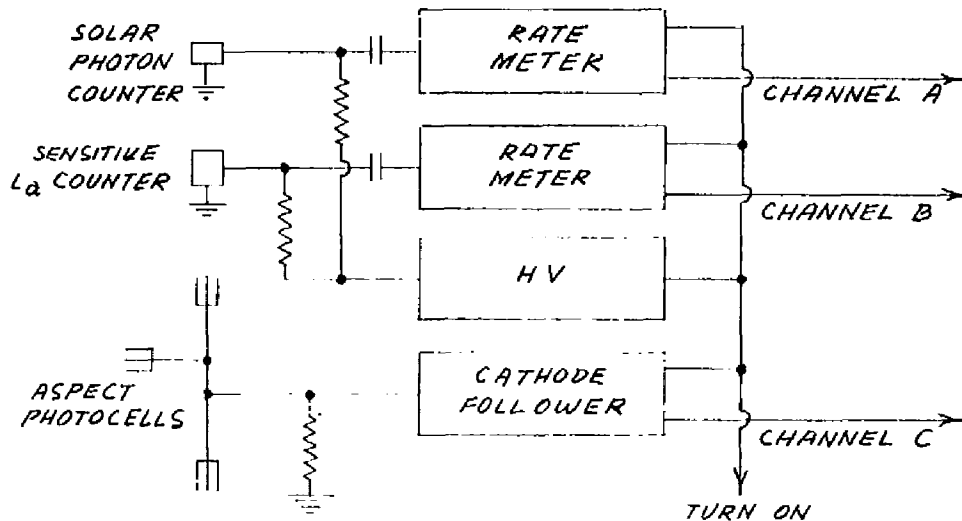


Fig. 18 - Satellite circuitry for determination of spatial hydrogen distribution

to detect the earth's albedo and a broadly collimated low-sensitivity photocell to provide aspect relative to the sun. Signals from the photocells are modulated by the spin of the satellite, and from the resulting amplitudes and phase relations the orientation of the detector axes relative to the earth can be determined. Since the information is generated by the spin of the satellite, it is desirable that the detectors' axes be perpendicular to the spin axis.

Any set of signals received from the system corresponds to eight possible orientations of the satellite. This ambiguity can be removed with the addition of three bits of information, i.e., the roll sense, nose up or down relative to the earth, and sun fore or aft relative to the spin axis. A schematic diagram of a proposed collimation coding is given in Fig. 19. A one-roll scan of a typical single-channel telemetry input is depicted in Fig. 20. The Lyman-alpha detectors are, of course, themselves sensitive to the sun and can yield the additional information required to give the roll sense.

This system has some blind spots, i.e., when the sun is near or below horizontal or near zenith, but it should give aspect adequate for our purposes for over 80 percent of the sunlit orbit. The aspect obtained during the daylight orbit may be followed through

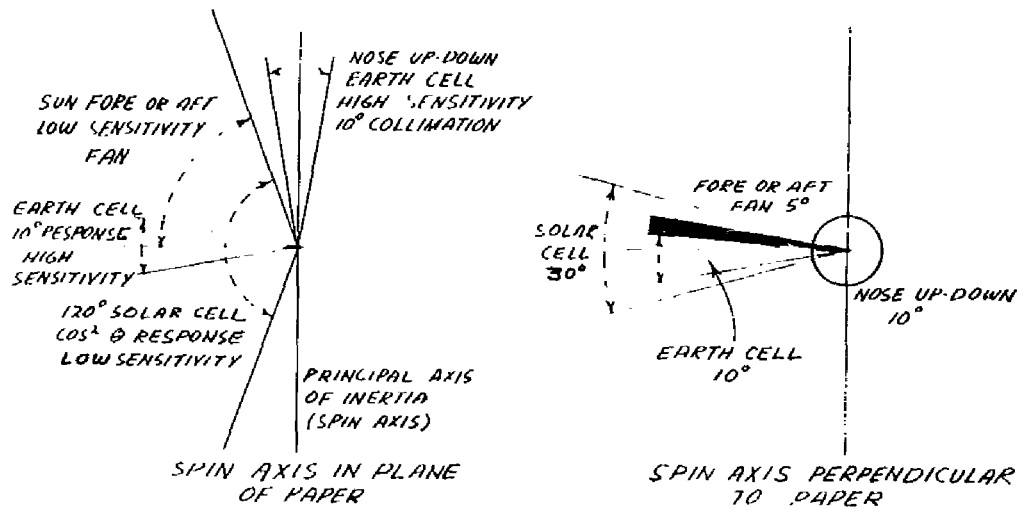


Fig. 19 - Collimation coding

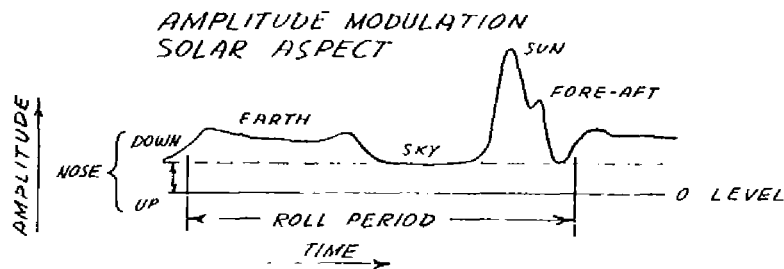


Fig. 20 - One-roll scan of a single telemetry input

the night portion by correlation with Lyman-alpha signals resulting from earth shine, the Milky Way, and other celestial sources.

#### 2.3.3.2 Satellite Exposure of Insensitive Nuclear Emulsions for Investigation of the Heavy Primary Nuclei of the Cosmic Radiation (Unclassified)

Nine years ago it was discovered that the primary cosmic radiation consists not of protons alone, but of heavier nuclei as well. Since that time the flux, chemical composition, and energy spectra of the heavy nuclei have been under investigation. However, we have only begun to elucidate those properties which can yield invaluable information

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on the origin of the cosmic radiation, its mode of acceleration, and various characteristics of the regions of space through which the radiation passes in traveling to the earth.

Among the problems which remain to be solved is a precise determination of the relative abundances of the various elements. Of special importance would be knowledge regarding the presence in the primary beam of elements heavier than iron, and a more unambiguous determination of the relative abundance of lithium, beryllium, and boron. Answers to these questions would have significant implications for the cosmological problems mentioned above.

The method proposed for this investigation involves the exposure of relatively insensitive nuclear emulsions. These could be exposed in a satellite for a period of, say, several days to a month. Re-entry through the atmosphere and recovery of the emulsion stack would be required. The weight of emulsion required for a good experiment depends on the duration of the satellite's flight. It could be reduced to a lower limit of about 5 pounds, with increased duration of flight. A desirable weight would be about 20 pounds for the emulsion stack alone. There is reason to believe that the weight of auxiliary equipment required for recovery would be at least 20 pounds. Accordingly, the use of a satellite with a payload of about 40 pounds would probably suffice. An orbit conforming as closely as possible to the plane of the geomagnetic equator would be desirable.

### 2.3.3.3 Satellite Studies of Certain Celestial Objects (Unclassified)

Ultraviolet radiations from stellar objects have been detected from rockets to an extent that has aroused great interest among astrophysicists and atmospheric physicists. Further investigation could be carried out more effectively from satellites than from rockets because of the longer observing times and the absence of the earth's absorbing and turbulent atmosphere.

Hydrogen has been detected in interstellar and interplanetary space by extreme ultraviolet and microwave techniques. A proposal was originally made for the Vanguard

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program to measure the density of hydrogen in interplanetary space by comparing Lyman-alpha radiation received from space with that emitted by the sun. The method is inherently capable of detecting the emission of hydrogen proton streams in space as well as the general background distribution. With additional weight capacity it would be possible to include continuous measurements of the contour of the solar Lyman-alpha line and any temporal variation in the depth of the core of the line. This latter measurement can be made with a grating instrument coupled to a photon counter detector. It would give a measure of the total hydrogen content between the satellite and the sun and information about the temperature of interplanetary hydrogen.

A very effective method of making ultraviolet surveys of the sky from a satellite would be by means of modified television. Sections of the sky would be scanned by telescopes with television type receivers sensitive to broad regions of the ultraviolet, say, 3000-2000A, 2000-1000A, 1000-100A, and selected line radiations. The results of these surveys would be used in planning investigations of particularly interesting celestial objects in experiments to be performed from manned and unmanned satellites.

An earth satellite affords the opportunity to measure the zodiacal light over a large range of angle with respect to the sun, and thus to bridge the gap in the distribution of energy in the zodiacal light from the solar F-corona to the "gegenschein." This experiment would need a satellite heavy enough to carry a sun-pointing control. By continuing the measurements over a year, the distribution of debris in the entire planetary system would be mapped out.

A study of the ultraviolet albedo of the moon may give additional information regarding the character of the moon's surface. This experiment would use a pointing control and spectrograph with photoelectric detector to give a measure of the solar ultraviolet radiation reflected from the moon. Viewing the sky close to the moon from different angles might provide also a means of detecting a possible trace of atmosphere by observing any luminescence of the atmosphere. These three factors, i.e., the ultraviolet

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radiation field at the lunar surface; the nature of the moon's crust; and the type of atmosphere, if any; are of vital importance to lunar landing parties from the earth.

Similarly, the ultraviolet albedos of the nearer planets would be of great aid in determining the properties of the planetary atmospheres.

### 2.3.3.4 Satellite Atmospheric, Particle, and Radio Astronomy Studies (Unclassified)

The experiments listed below are not discussed in as much detail as are many of those in the foregoing sections. Nevertheless, they are generally based on proved techniques and sound principles. They are examples of the many possibilities which can be exploited once adequate satellite vehicle capabilities are available.

#### 1. Micrometeor experiment.

Purpose - to measure micrometeor fluxes and penetrating power.

Method - cadmium sulphide cells covered by different foils. Total weight 20 pounds, total volume 0.3 cu ft.

This experiment could be ready in about 6 months.

#### 2. Micrometeor latitude variations

Purpose - to measure latitude variation of micrometeors.

Method - use microphone and photopulse detectors.

20 pounds, 0.25 cu ft.

This experiment could be ready in about 1 year.

#### 3. Cosmic ray experiment.

Purpose - to measure cosmic ray composition, origin, and reasons for fluctuations.

Method - counters 20 pounds, 0.25 cu ft.

This experiment could be ready in about 1 year.

#### 4. Cosmic ray latitude effect experiment.

Purpose - to determine spatial and temporal variation in the "knee"

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for different cosmic ray components in polar orbit  
satellite. 40 pounds.

This experiment could be ready in about 1-1/2 years.

5. Lunar ionosphere experiments

Purpose - to measure positive ion composition between  
the earth and the moon.

Method - Radio frequency mass spectrometer. 40 pounds,  
.0.3 cu ft.

This experiment could be ready in about 1-1/2 years.

6. Earth's atmosphere as quasi-luneberg lens.

Purpose - use earth's atmosphere as quasi-luneberg  
lens for radio astronomy.

Method - place satellite in orbit at about 5000 miles,  
containing radiometer and simple feed horn or dipole  
array. If orbit can be made to precess slowly, entire  
celestial sphere can be covered. Operating frequency  
would probably be in UHF or VHF. Weight, 50 to 100 pounds;  
power, perhaps as low as 50 watts; both exclusive of telemetering.

7. Galactic background temperature experiment.

Purpose - measure low-frequency temperature of galactic  
background.

Method - use 50-kc receiver with trailing wire or loop to  
detect high-temperature radiation from Milky Way and  
other sources, in satellite orbit of at least 1000 miles  
altitude. Weight, 30 pounds; power 25 watts; both exclusive of  
telemetering.

## 2.4 MANNED FLIGHT AND BIOLOGICAL EXPERIMENTS (CONFIDENTIAL)

The essential biological researches leading to manned flight can all be completed by conducting an appropriate set of experiments with chimpanzees. Such a set of experiments can be in the medium-sized satellites which can be launched by means of Thor-Vanguard combinations. These experiments would probably be conducted in satellites with 30-degree orbital inclinations. The complete capability for conducting these experiments rests within the Naval establishment. The next step, manned flight, will require an extensive testing program which will lead to the required degree of reliability. These tests will involve larger satellites which can be recovered. It appears that a number of important lunar experiments can be profitably conducted in connection with these tests.

## 2.5 SATELLITE DEVELOPMENT (UNCLASSIFIED)

The satellite's electric field can have an important effect upon equipments within it. This field is, in turn, a function of the ambient electron temperature. Accordingly, measures of these two quantities are fundamental to certain satellite applications. These measures, and the stability and recovery problems, are discussed in the following sections.

### 2.5.1 SATELLITE CHARGE AND INTERPLANETARY ELECTRIC FIELD EXPERIMENT (UNCLASSIFIED)

The purpose of this experiment is to measure the interplanetary (terrestrial, lunar, and solar) electric field; and the satellite's electric charge, potential, and sheath thickness.

The method would be to use two symmetrically-located electric field meters to separate the ambient field from the field due to the satellite charge. New field meters two orders of magnitude more sensitive and stable than those flown in V-2 rockets since 1948 will be available within 6 months. These instruments can measure the electric field independently of the photoemission and conduction currents, which flow as a result of the high conductivity of the ionized low-density environment. Satellite spin can be used to advantage, particularly in the earth's shadow, to obtain accurate values of ambient field down to about 0.1 v/m. A retarding-potential experiment (or an ion mass spectrometer) can be used to determine the potential of the satellite. The space-charge sheath thickness is then determined from the surface potential gradient and the total voltage drop (satellite potential) across this sheath. The equipment would weigh about 30 pounds and occupy a volume below 0.3 cu ft. This experiment can be ready in about 1-1/2 years.

### 2.5.2 SATELLITE CHARGE MEASUREMENTS UTILIZING AN ARTIFICIAL CHARGING MECHANISM (UNCLASSIFIED)

Under any given set of environmental conditions, the satellite would be expected to approach an equilibrium potential. If this potential is perturbed, information concerning the environment might be obtained by measurement of the rate of decay back to the equilibrium potential. Determinations of satellite potential would be made by means of a generating voltmeter as suggested in the preceding proposal, if it is assumed that the

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potential changes are not too rapid. The device proposed to change the charge on the satellite is an electron gun. It could be operated either in bursts to cause a transient potential change, or continuously to maintain an altered satellite potential. In the latter case the measurement of interest would be the ejected electron current necessary to maintain this potential.

The electron gun itself could be small and light (weighing less than a pound). The electrical power requirements would be small for pulse operation at low repetition rates, but might become prohibitive for continuous operation if the environmental electric currents are large.

### 2.5.3 ELECTRON TEMPERATURE (UNCLASSIFIED)

The purpose of this experiment would be to measure electronic temperature at satellite altitudes. This measurement is independent of the charge on the satellite.

The method would employ an open photomultiplier tube. The weight would be 20 pounds, and the volume 0.25 cu ft. This experiment could be ready in about 1 year.

### 2.5.4 A DARK SATELLITE (UNCLASSIFIED)

Because of visibility considerations, the Vanguard satellites have specular reflecting surfaces. All problems of development of surfaces, including mechanical, optical, and thermal control problems, have been centered around this requirement for high specular reflection. The concept of a black-surfaced satellite leaves some problems unchanged and poses others which require study and investigation.

The temperature problem is only slightly different from that of the Vanguard Group I (NRL Lyman-alpha) satellites. The same basic calculations are to be used, and the major controllable parameter is  $a/e$ . For a black surface,  $a \approx 1$ , and, if the equilibrium temperature is to be similar to that of the Vanguard satellites,  $e$  must be made nearly unity. The ultimate equilibrium temperature uncertainty will be  $\pm 30^\circ \text{C}$  as for the Vanguard Group I satellites, but the temperature excursions of the shell may be greater than the  $\pm 20^\circ \text{C}$  predicted for the Group I units. These excursions will depend on the absolute values of

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a and e and on the heat capacity of the shell. The uncertainty range of  $\pm 30^{\circ}\text{C}$  in equilibrium temperature may be decreased on the basis of information from successful Vanguard satellites. In general, therefore, the concept of a black coating poses no new problems in the realm of temperature control.

There are some mechanical problems introduced however. For example, the thermal properties of a black coating must be satisfactory. The coating must stand temperatures of up to  $150^{\circ}\text{C}$  caused by in-flight heating. The coating must be adherent despite repeated temperature excursions of from  $40^{\circ}\text{C}$  to  $100^{\circ}\text{C}$  peak-to-peak. The stability of the coating under intense short-wavelength ultraviolet irradiation must be determined, and the effect of a dissociated- $\text{O}_2$  environment must be estimated. Finally, any proposed coating must be checked to determine that its infrared absorptivity is the same as its visible absorptivity. These problems are not major ones. A number of satisfactory coatings probably can be developed and tested in six months to a year.

### 2.5.5 GYROSCOPIC STABILIZATION OF THE SATELLITE (UNCLASSIFIED)

Many experiments possible in both the satellite and the moon rocket require attitude control. It is proposed therefore to develop a system employing a three-axis flywheel system (similar to a gyro system), in which deceleration or acceleration of the flywheels would cause the desired changes in attitude. Either of two attitude reference systems are initially proposed: Sun-sensing and orientation to the sun-fixed attitude, or earth-edge sensing and orientation in an attitude fixed with respect to the earth-vertical.

Assuming the use of NRL Engineering Service facilities, the first design of an attitude control servo for a satellite should be completed in six months to a year. It is initially feasible to use a commercial gyro adapted to solar cells for power, and modified to permit control of the driving motor. The weight of the package should be less than 10 percent of the total weight of the satellite and still permit control of attitude at the slow speeds required to maintain an attitude fixed by the sun or the earth.

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### 2.5.6 SATELLITE RECOVERY (CONFIDENTIAL)

The present Vanguard IGY satellite program calls for the use of small, thin-shelled spheres instrumented to study the upper atmosphere. The next logical step is to provide for the recovery of satellites. Such recovery is not only desirable but also essential for many reasons. First, there are numerous possible and valuable satellite experiments which would provide data unsuited to radio transmission. Second, certain types of photographic reconnaissance could be accomplished readily. Third, the trend is toward bigger and heavier satellites which, if not controlled, will sooner or later return to earth: if one of these should crash into a heavily populated area, it might do great damage. Fourth, one of the aims of the satellite program is to open the door to space flight, and one of the major obstacles to space flight is the problem of a controlled return to earth.

A recoverable satellite program could use either of 2 techniques: (1) natural recovery, and (2) forced recovery. Natural recovery is that in which the satellite's re-entry is due to natural forces, e.g., gravity and air drag, while forced recovery utilizes propulsion to force the satellite out of its orbit and into a re-entry trajectory. Both techniques will require considerable information from the present satellite program, particularly the information on atmospheric density, and the drag and heat-transfer coefficients for spheres. Furthermore, knowledge of the accuracy of the guidance and tracking systems is required for the projection of satellites into short-lifetime orbits and for the prediction of impact location.

At present, it appears most reasonable to begin the recoverable satellite program with a sphere. Regarding the first phase of the program, natural recovery, it is possible that the use of high-temperature alloys as structural material would permit the satellite to return to earth without burning up. Therefore the first step would be to build such a sphere capable of surviving the impact and capable of floating; this will be referred to here as Test Sphere A. Test Sphere A should also contain a beacon to permit its location after impact. Its surfact should be such as to secure minimum heat transfer rates, i.e., high emissivity, low accommodation, and should be noncatalytic, insofar as the compatibility

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of these requirements will allow. The use of ablation techniques may be required if the heating is severe. A method of orienting Test Sphere A during its final dive will also be required in order that the beacon antenna remain above ground after impact. Furthermore, reverse thrust rockets might be utilized in the high-heating region if orientation is accomplished early enough. Such early orientation might be accomplished in one or a combination of several ways, e.g., an afterbody sufficiently large to make the body reasonably stable aerodynamically, or the use of small wings and/or a parachute (particularly if impact conditions are too severe for the structure or beacon). Means of telemetering heat transfer data and lift forces should be included. It is conceivable that the information required and the weight allowed for Test Sphere A will be incompatible. Therefore Test Sphere A might become a series of test spheres each concerned with a particular part of Phase 1, the starting of natural recovery techniques. Eventually, Phase 1 will have to include the use of wings and variable lift in order to obtain reasonable control over what will be essentially a recoverable satellite glider. All Phase 1 satellites would be short-lifetime satellites (i.e., about 10 revolutions) with orbits as nearly circular as possible. Concurrent with Test Sphere A, other shapes should be studied to determine their suitability with regard to drag and heat-transfer coefficients.

Phase 2, forced recovery, presents a more difficult task. Forced recovery is envisioned as involving a thrust opposed to the direction of the satellite's velocity vector to decelerate the satellite so that gravity and drag will bring it down at a desired point. This technique requires that the orientation of the satellite with respect to the earth be either known or controlled.

## 2.6 SATELLITE SYSTEMS (SECRET)

The satellite uses described above seem to fall into two categories insofar as the orbit is concerned. The reconnaissance, infrared, and optical detection satellites seek targets in the Soviet Union. An orbital inclination of about 60 degrees seems best for these satellite applications. A satellite weighing at least 300 pounds could conduct several of the types of reconnaissance described above. Ideally, it would include all which have a bearing upon a single phase of Soviet operations, since correlation is always of great importance in intelligence estimates. For example, a satellite vehicle designed to reconnoiter the Soviet atomic testing grounds should include not only the nuclear detection equipment but also appropriate electronic intelligence equipment for detecting any correlated electronic transmissions associated with the nuclear testing operations (Fig. 20 and Table 7). Similar remarks apply to the combinations of infrared and optical ballistic missile detection systems, and electronic ferrets designed to reconnoiter radiations associated with the Soviet missile testing ranges (Figs. 21 and 22).

For the same reasons, satellites including combinations of all these types of reconnaissance systems (Fig. 23) would be most useful. Such satellites should be silent over Soviet territory. Transmissions to the ground receivers should be made at locations under U. S. control which are as well hidden as possible. Shifting of the radio frequency would make it more difficult for clandestine listeners near the receiving points to attempt to determine the orbits of such satellites from Doppler observations. These satellites should also be invisible to the naked eye. Satellites of moderate size could be rendered effectively invisible by means of blackened or solar cell surfaces. Heat switches or cold spots would probably make it possible to keep the satellite temperature within proper bounds. Such satellites, if launched under proper conditions, would be exceedingly difficult for the Soviets to find.

Just three such satellites would make it possible to reconnoiter key Soviet locations such as Moscow and the missile and atomic testing ranges for an interval of the order of a tenth of a period each period. This would not provide complete coverage. However, if

TABLE 7  
Reconnaissance Satellites  
300 lb, 60° Orbit, Dark

1. Atomic Test Reconnaissance  
Thermal, neutron and radioactivity measurements, electronic intelligence, and TV
2. Missile Test Reconnaissance  
Infrared and optical measurements, electronic intelligence, and TV
3. Electronic Intelligence and TV - Moscow Complex

the presence of these satellites were not suspected, or even if their ephemerides were not accurately known to the Soviets, a reasonable amount of surveillance information could be obtained. If the Soviets knew of the presence of these satellites and suspected their nature, they would probably try to build their countdowns around the orbits, scheduling the missile launchings or atomic shots at times when the satellites were out of sight. This would constitute a harassment. If it were deemed worthwhile, complete coverage could be provided by putting up several dozen satellites.

Many of the satellite systems described above could take advantage of polar orbits. This is true, for example, of the navigation systems, the radioactive sampling systems, and many of the geophysical research satellite systems (Figs. 24 and 25, and Table 8). Still others such as those involving certain types of solar studies or cosmic-ray studies could utilize orbits of the type now planned for the Vanguard satellites.

These latter satellites and the polar satellites could form a set of announced U. S. satellites. A few announced satellites might even be launched along 60-degree orbits, to help confuse the Soviets.

The polar satellites and those having orbits whose inclination is 60° would have to be launched from a west coast location such as the Camp Cooke Interim Operational Capability Site now planned by the Air Force. The 60 degree orbits would probably actually be retrograde orbits. It would probably be desirable for several reasons to include the scientific experiments in the polar satellites or those launched along orbits of the type now planned

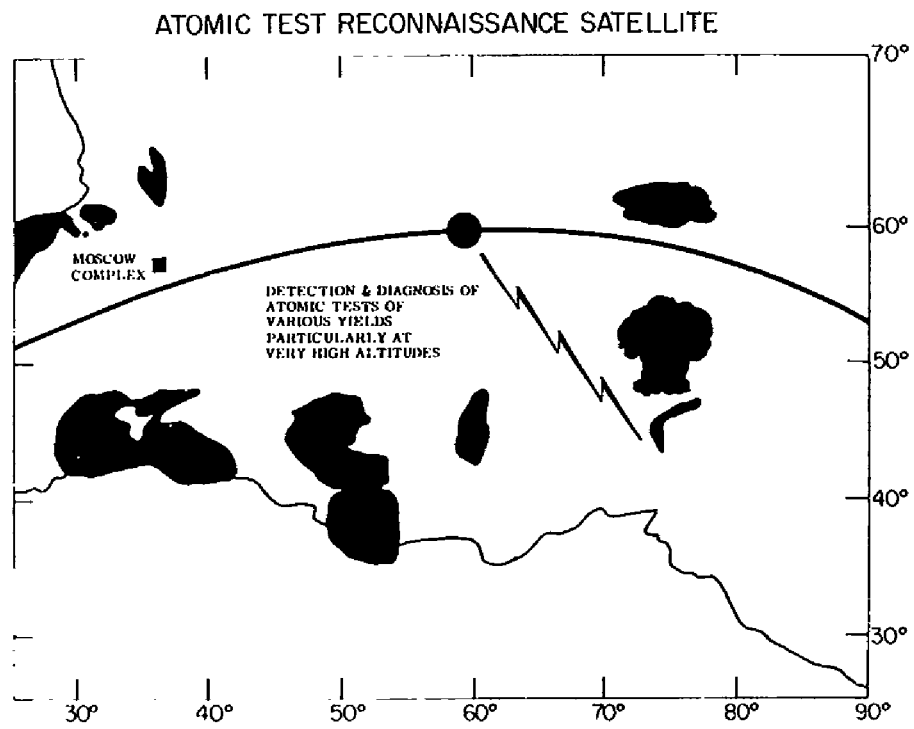


Fig. 20

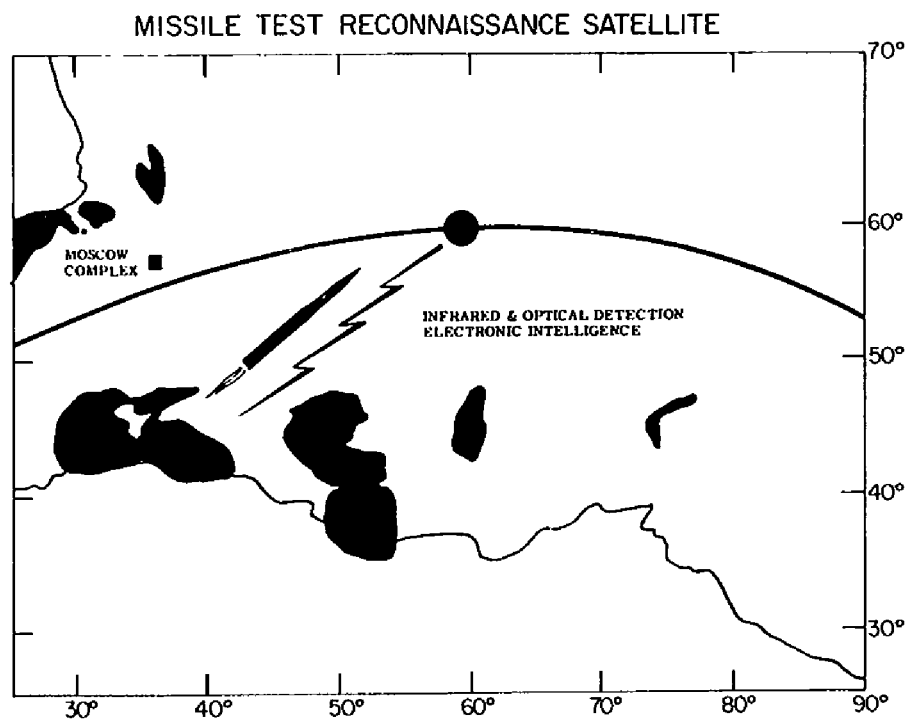


Fig. 21

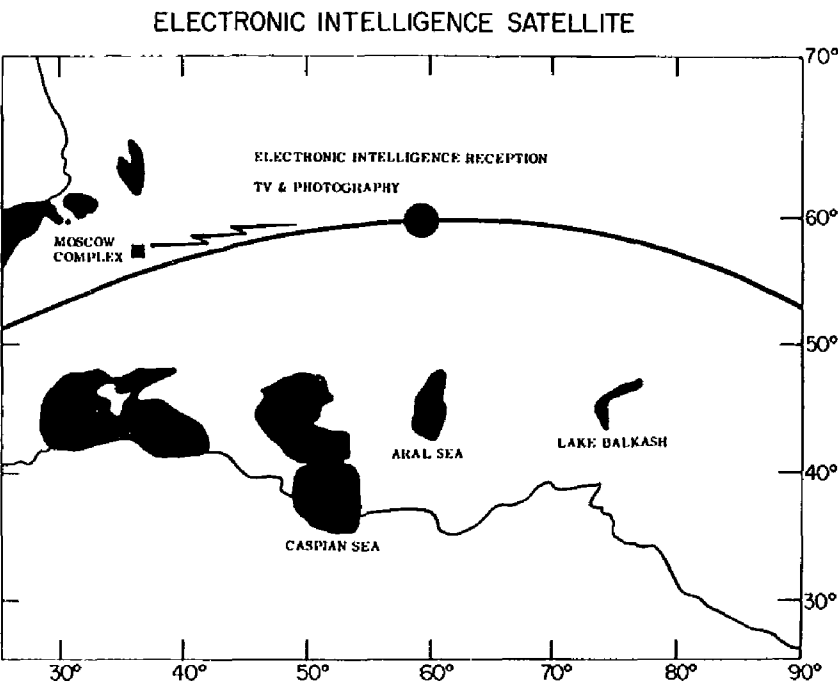


Fig. 22

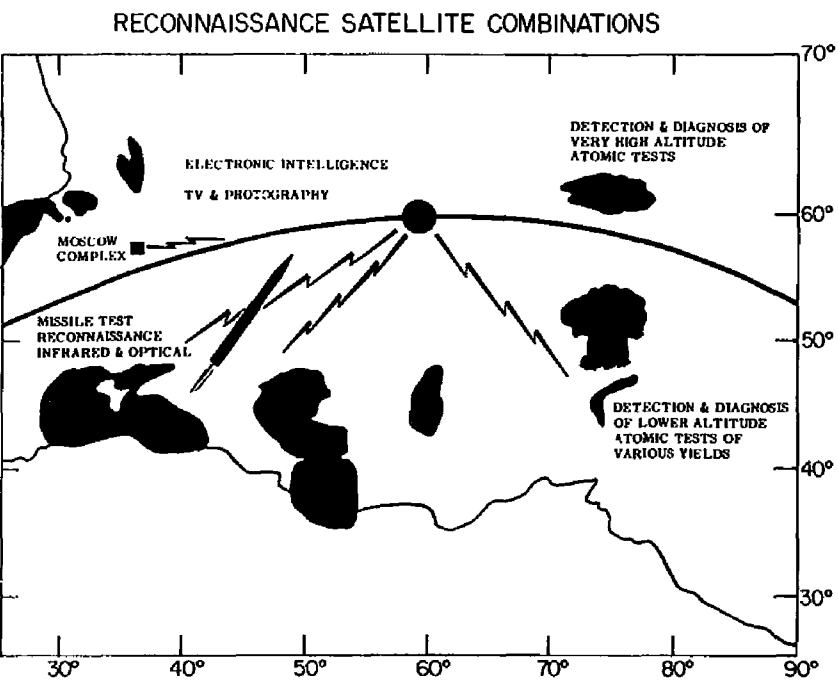


Fig. 23

POLAR NAVIGATION COMMUNICATION AND RECONNAISSANCE  
DARK SATELLITE COMBINATION

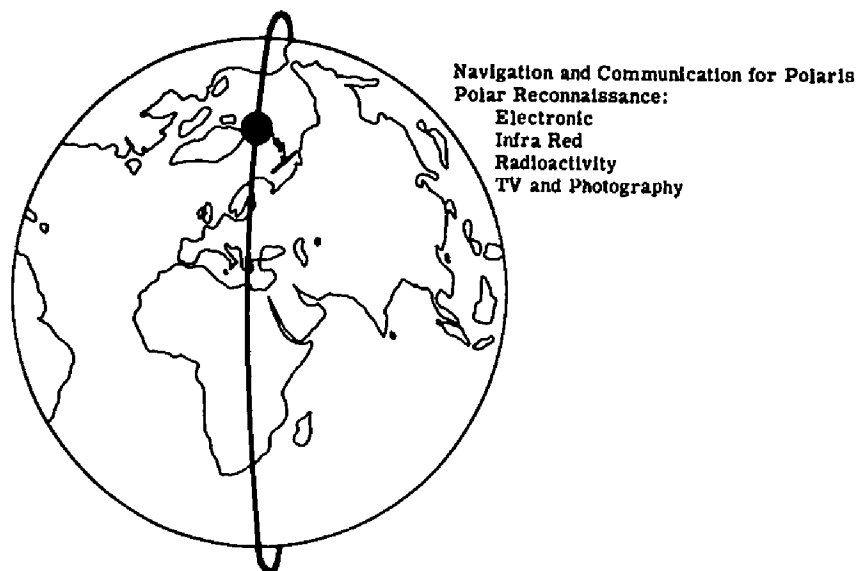


Fig. 24

NAVIGATION, WEATHER, AND SCIENTIFIC  
SATELLITE COMBINATION

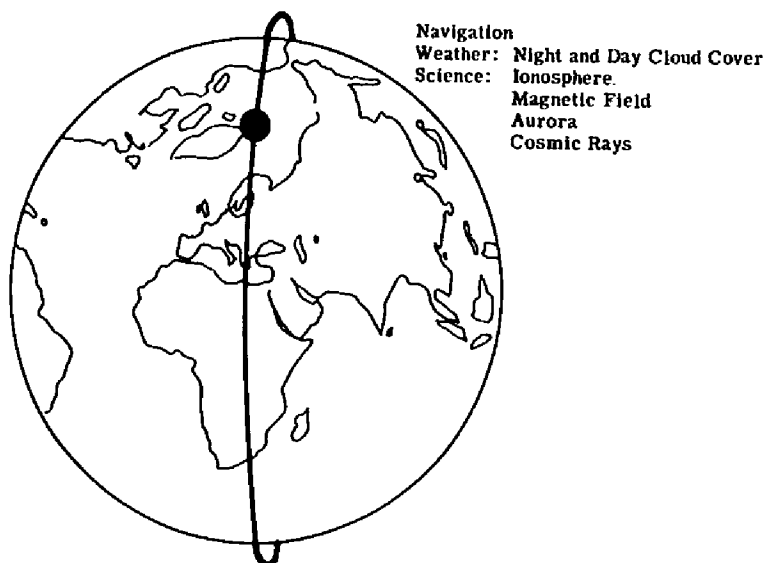


Fig. 25

TABLE 8  
Navigation, Communication, Reconnaissance  
and Scientific Satellites  
300 lb, Polar Orbit

1. Dark - Navigation and Communication for Polaris,  
Polar Reconnaissance: Electronic, I-R, TV,  
Radioactivity.
2. Announced - Navigation, Night and Day Cloud Cover,  
Radioactivity,

Scientific Exp'ts: Ionosphere, Magnetic Field,  
Aurora, Cosmic Rays, Etc.

for the Vanguard satellites. The publication in unclassified literature of researches conducted by means of these satellites would involve reference to the orbits. If these orbits were those of reconnaissance satellites it would be difficult either to make proper reference to them or to maintain a maximum effort to keep the knowledge of the existence of these orbits classified.

## CHAPTER 3

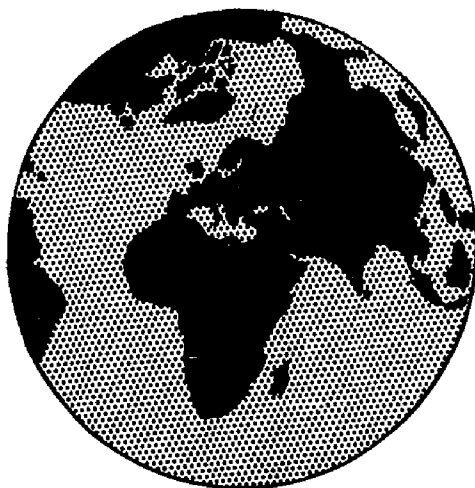
# PIONEERING LUNAR VEHICLES

### 3.1 INTRODUCTION (CONFIDENTIAL)

Pioneering lunar vehicles fall into several categories. They may actually strike the moon, may circumnavigate the moon, or may become artificial satellites of the moon. Those which strike may impact at the full velocity, or they may be landed at a speed which would permit equipment to operate after arriving at the moon's surface. These are sometimes referred to as hard and soft landings, respectively.

It has been thought that the designer of a pioneering lunar vehicle payload faces a fundamental problem: he must decide whether the vehicle is expected to hit the moon or miss the moon. Some payloads are appropriate to the former case, others, to the latter. An instrument for measuring the moon's magnetic field is a happy choice for a lunar vehicle payload since it does its work whether or not the moon is actually struck by the payload (Fig. 26). As long as the lunar vehicle comes within a reasonable distance of the moon, it can make its measurement. If it impacts the moon, an important part of its work will already have been done. Possible payloads for pioneer lunar vehicles are discussed in the following sections.

#### PIONEER MOON ROCKET



#### LUNAR MAGNETIC FIELD MEASUREMENTS



#### LUNAR RADIO BEACON

Fig. 26

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### 3.1.1 MAGNETIC FIELD MEASUREMENTS FROM AN EARTH-TO-MOON VEHICLE

#### 3.1.1.1 Objectives (Confidential)

The objectives of these experiments would be the measurement of (a) the moon's magnetic field, (b) interplanetary (or interstellar) magnetic fields, (c) the strength and location of a permanent ring current surrounding the earth, and (d) the drop-off of the earth's field at large distances.

#### 3.1.1.2 Primary Reasons For Pursuing These Objectives (Confidential)

Knowledge of the moon's magnetic field might provide the additional clues necessary to explain the source and origin of the earth's magnetic field. It is interesting to note that both the liquid-core dynamo-current theories and the permanent magnetization theories assume an initial field of unknown origin. If a similar initial field existed in the moon, the moon's field could approach the earth's field in magnitude. It is likely that a significant lunar magnetic field would delay the escape of the ionized lunar gases, thus causing a more dense lunar "ionosphere" than would otherwise be expected. Magnetic field measurements would also provide information on the internal constitution of the moon, its mineral content, etc.

Magnetohydrodynamic theories and studies of the polarization of starlight both indicate the presence of interplanetary and/or interstellar magnetic fields. Estimates of field strengths vary over a wide range, approximately  $10^{-3}$  to  $10^{-8}$  gauss. Knowledge of the true strength would apparently be useful in radio astronomy as well as cosmology.

Permanent (or nonvariant) ring currents about the earth at distances of several or more earth radii are postulated in many prominent theories (e.g., Chapman, Alfven, Bennett) of geomagnetism, magnetic storms, and aurorae. It is possible that such currents contribute as much as 1 percent to the fields at the earth's surface and much larger percentages at extreme altitudes. The importance to the understanding of particle bombardment of the earth is obvious. Field changes amounting to 5000 gammas or more might be encountered in traversing such a ring.

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Measurements of the magnetic field at large distances from the earth would give a sensitive value for the earth's dipole moment. From the decrease of the field with altitude, information on the quadrupole and higher order moments would also be obtained.

### 3.1.1.3 Range of Field Intensities (Confidential)

A study is in progress to determine the most probable field intensities for the moon, interplanetary space, and a permanent ring current. Estimates can vary over wide ranges depending on the assumptions made. One point that is apparent is that it would be wise to have an instrument which could make measurements in fields as weak as one gamma ( $10^{-5}$  gauss) and as strong as 0.1 gauss.

### 3.1.1.4 Instruments (Confidential)

The only magnetometer now known which is capable of absolute measurement of very weak fields is the alkali vapor resonance magnetometer proposed by Varian Associates. This instrument has the additional advantages that measurements are independent of orientation and electronic drifts. Estimates indicate an instrument weight of one pound (excluding batteries) and a power dissipation of one watt. With Zn-AgO batteries (50 watt-hours per pound), a pound of batteries would be required for each two days of continuous operation. For long lifetimes this power could be supplied by reasonably small areas of silicon solar cells. Telemetering requirements will depend greatly on the relaxation time of the particular alkali vapor used as well as its resonant frequency. High frequencies (100 kc to 1 Mc) can probably be counted and read-out with a small weight of magnetic-core counters, already available. At lower frequencies direct transmission or beat-frequency techniques could also be used. As telemetering power will be a prime consideration, probably requiring intermittent operation, it is likely that the modulation technique will be dictated by the telemetering system.

If the experiment is purely exploratory to detect the presence of magnetic fields and not to measure their magnitudes accurately it is definitely feasible to use a component saturable core magnetometer similar to the rocket aspect units used at NRL. Fields as

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low as 1 gamma would be detectable from the spin modulated output caused by the changing orientation of the sensor. Placing the emphasis on spin modulation removes the drift problems likely to be encountered in this type of magnetometer. A logarithmic type of output would probably be required to detect strong fields as well as fields as low as one gamma. The instrument weight would be about one pound. The power required would be about 0.1 watt. The telemetering modulation could be dc or ac depending on the response time, the spin rate, and the importance of getting a rough measure of field magnitude as well as its presence.

### 3.1.1.5 Estimated Development Time (Confidential)

With sufficient support, a prototype alkali vapor resonance magnetometer for use in an earth-to-moon vehicle could probably be produced in six to nine months. A period of twelve to fifteen months is estimated for the completion of tested flight units. A saturable core magnetometer for this application could probably be produced in six to eight months. Developments in addition to either of the magnetometers will depend almost entirely on the telemetering input requirements.

### 3.1.2 PROOF OF A LUNAR ACHIEVEMENT

#### 3.1.2.1 An Artificial Lunar Satellite (Confidential)

Proof of a lunar achievement can be attempted in various ways. The one method for doing this would be to establish an artificial satellite of the moon. A 12-foot sphere would be about a 17th magnitude object at the moon's distance. Such a large satellite system is described in Chapter 2 of this report. It is estimated that a lunar satellite system weighing approximately 50 lb could be built to include such a balloon, a radio transmitter capable of being received by the earth-based Minitrack System, instrumentation for measuring the lunar magnetic field, and a Vanguard integrating accelerometer for effecting velocity cutoff.

#### 3.1.2.2 Proof of a Moon Impact (Confidential)

If the pioneering lunar vehicle actually strikes the moon, it will be of paramount interest to prove it. Telemetered signals from a magnetometer, correlated with trajectory information, could probably provide proof which would be reasonably convincing to those who were familiar with this type of evidence. The target here is not this audience, however.

Proof of such a feat could be provided by a radio beacon planted on the moon. Such a beacon could be powered by batteries, solar cells, or both. Such a beacon would also be of definite value as a navigational aid. It would not exhibit the jitter in position that sometimes occurs in a signal reflected from the moon. This lack of jitter would add to the proof that the moon had actually been impacted. For the accurate use of this beacon, only a moderate antenna gain and an ordinary receiver would be required. At a frequency of, say, 400 Mc, a power level of 20 watts would provide a usable signal with a ground antenna gain of 100, which could be obtained from a 16-element array with dimensions of about 8 by 8 feet. Provisions would be necessary to prevent the beacon from burying itself in the pumice or dust on the moon's surface, and to allow its solar-cell power unit to be irradiated by the sun. The use of retro-rocket and spike landing equipments would be

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called for. Such a system would be fairly sophisticated relative to the immediate state of the art. A desirable additional feature of the beacon would be the use of a crystal-controlled oscillator with a quartz crystal of a known frequency-temperature characteristic, so that the moon's surface temperature could be obtained from the signal. In the interest of identifying the signal, it would be worthwhile to consider the use of a small coder to transmit the letter A for American (.-) once every few seconds.

## CHAPTER 4

# SATELLITE LAUNCHING SYSTEMS

### 4.1 SATELLITE LAUNCHING VEHICLES

#### 4.1.1 INTRODUCTION (SECRET)

At least two satellite launching vehicle systems are being considered for the immediate post-Vanguard period. The first type is an improved Vanguard vehicle. It differs from the present vehicle chiefly in that the third stage has a higher performance. This improved Vanguard vehicle should be capable of putting 55 pounds in an orbit of the present type, and about 35 pounds in a polar orbit. The second type of vehicle is a combination involving the Thor as the first stage and the Vanguard second and third stages as its second and third stages. This vehicle should be capable of putting a 350-pound satellite into a polar orbit. Other types of vehicles are also being considered. Each of these involves the use of the Thor IRBM as the first stage. Each also involves the use of a larger version of the present Vanguard second stage, delivering approximately twice the total impulse of the present second stage. One version includes a cluster of three of the high-performance Vanguard third stages as its third stage. Another version includes a fourth stage consisting of a single high-performance Vanguard third stage.

These combinations are reasonably well optimized, at least insofar as the top stages are concerned. Owing to a Thor-imposed limitation, the total weight of the top stages and the payload cannot exceed 10,000 pounds. Accordingly each of the vehicles under consideration is tailored to meet this requirement. The use of these larger top stages makes it possible to increase the payload by approximately a factor of 2.

The use of these larger top stages would, of course, involve additional problems. Only a partial analysis has been made of these systems.

Combinations of the Thor and the Vanguard top stages do, however, offer real promise of providing the country with the capability of putting several hundred pounds in orbit at the earliest possible moment. The growth potential of this system also appears to be excellent.

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Characteristics of these systems are shown in the following pages. The principal characteristics are shown in Table 9.

**TABLE 9**  
**Estimated Payload Capabilities of**  
**Improved Vanguard and Thor-Vanguard**

| Vehicle                | No. of Stages | Payload Weight (lb) |             |                |             |
|------------------------|---------------|---------------------|-------------|----------------|-------------|
|                        |               | Camp Cooke          |             | Cape Canaveral |             |
|                        |               | 60° Orbit           | Polar Orbit | 30° Orbit      | Lunar Orbit |
| Improved Vanguard      | 3             | 25                  | 35          | 55             | -           |
| Thor-Vanguard          | 3             | 300                 | 350         | 475            | 50          |
| Improved Thor-Vanguard | 3             | 500                 | 600         | 800            | -           |
| Improved Thor-Vanguard | 4             | 600                 | 700         | 900            | 125         |

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4.1.2 THE IMPROVED VANGUARD VEHICLE (SECRET)

The improved Vanguard vehicle is shown in Fig. 27. Its principal characteristics are shown in Table 10 and Figs. 28 and 29.

TABLE 10  
Improved Vanguard Vehicle Characteristics  
Projection Velocity: 27,000 fps  
Projection Altitude: 200 mi

| Stage Type  | <u>1st Stage</u><br>Vanguard | <u>2nd Stage</u><br>Vanguard | <u>3rd Stage</u><br>Vanguard |
|---|------------------------------|------------------------------|------------------------------|
| Total Impulse (lb-sec)                              | 3,899,127*                   | 888,195†                     | 99,750 †                     |
| Specific Impulse (sec)                              | 250*                         | 274†                         | 265 †                        |
| Thrust (lb)   | 27,835*                      | 7,700†                       | 2,850†                       |
| Burning Time (sec)                                  | 140.08                       | 115.35                       | 35                           |
| Chamber Pressure (psia)                             | 616                          | 206                          | 215                          |
| Nozzle Expansion Ratio                              | 5.5:1                        | 20:1                         | 26:1                         |
| Total Impulse/Weight Ratio<br>Without Payload (sec) | 218.8*                       | 208.6†                       | 230.4†                       |
| Dry Weight Without Payload (lb)                     | 1,578                        | 914                          | 60                           |
| Usable Propellant Weight (lb)                       | 15,596                       | 3,241                        | 373                          |
| Payload Weight (lb)‡                                | 4,726                        | 468                          | 35                           |
| Total Takeoff Weight<br>With Payload (lb)‡          | 22,543                       | 4,726                        | 468                          |
| Total Takeoff Weight<br>Without Payload (lb)        | 17,817                       | 4,258                        | 433                          |
| Mass Ratio with Payload‡                            | 0.6918                       | 0.6858                       | 0.7970                       |
| Mass Ratio without Payload                          | 0.8754                       | 0.7613                       | 0.8614                       |
| Fuel Outage and Trapped (lb)                        | 642                          | 103                          |                              |

\*At sea level

†In vacuo

‡For polar orbit

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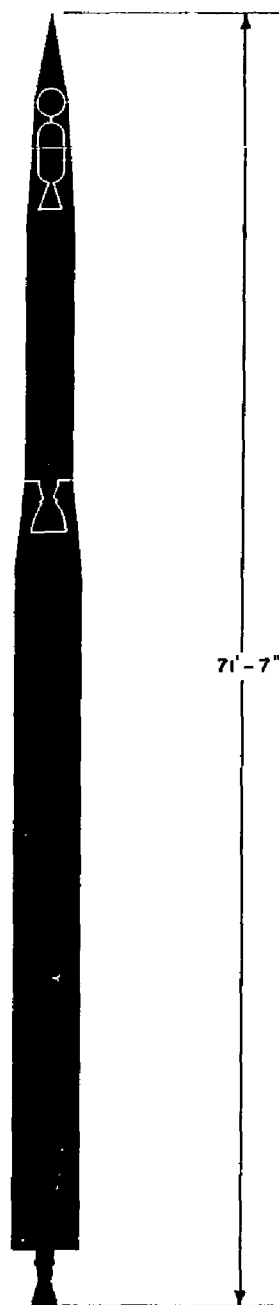


Fig. 27 - Improved Vanguard vehicle

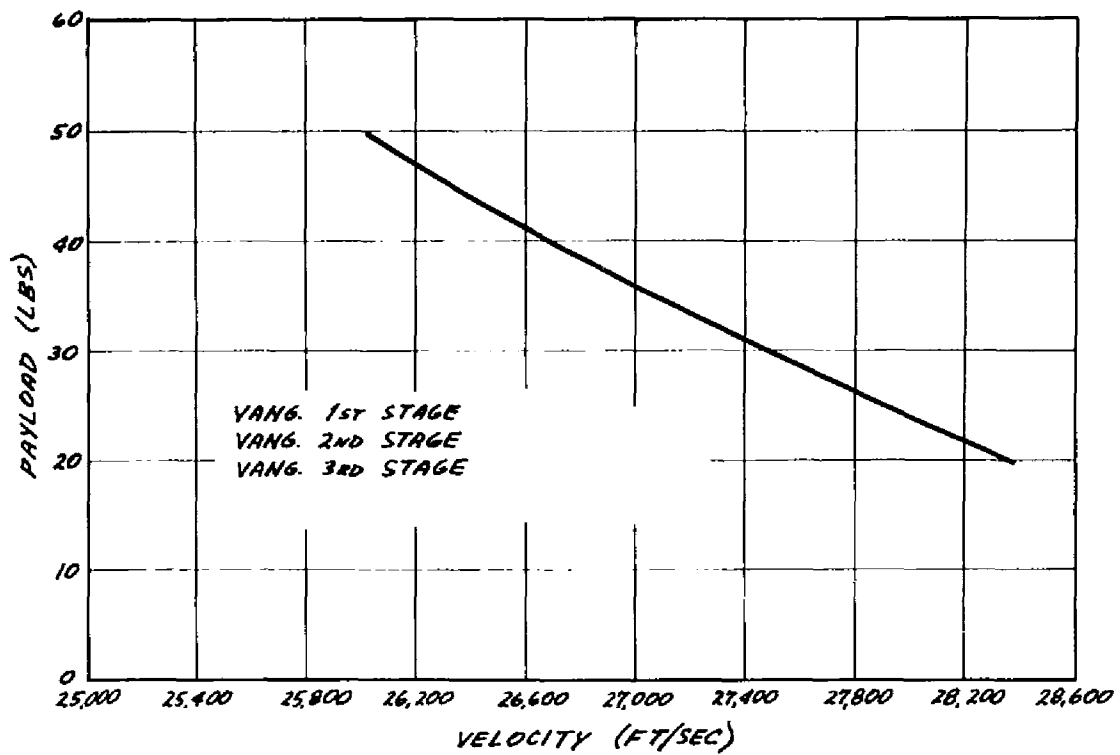


Fig. 28 - Performance of improved Vanguard vehicle for polar orbit

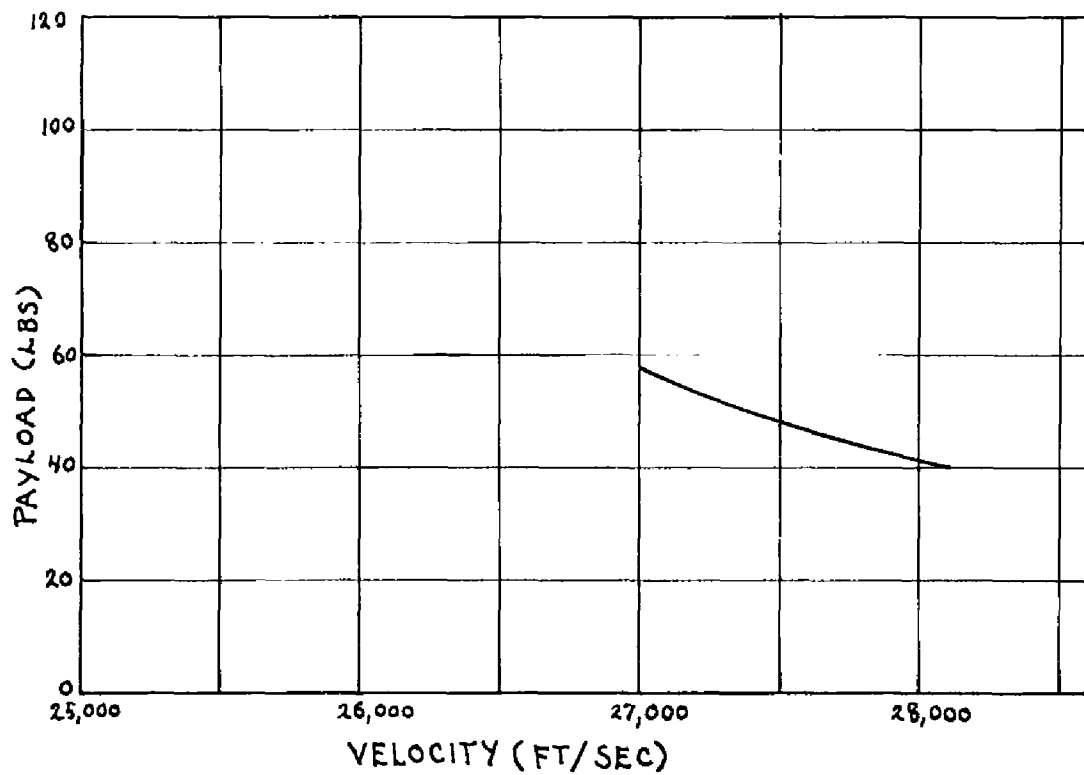


Fig. 29 - Performance of improved Vanguard vehicle for 35° orbit

## 4.1.3 THE THOR-VANGUARD VEHICLE (SECRET)

The Thor-Vanguard vehicle is shown in Fig. 30. Its principal characteristics are shown in Table 11 and Figs. 31, 32, and 33.

TABLE 11  
Thor-Vanguard Vehicle Characteristics  
Projection Velocity: 27,000 fps  
Projection Altitude: 200 mi

| Stage Type   | 1st Stage<br>Vanguard | 2nd Stage<br>Vanguard | 3rd Stage<br>Vanguard |
|--|-----------------------|-----------------------|-----------------------|
| Total Impulse (lb-sec)                                 | 24,159,300*           | 888,195 <sup>†</sup>  | 99,750 <sup>†</sup>   |
| Specific Impulse (sec)                                 | 250*                  | 270 <sup>†</sup>      | 265 <sup>†</sup>      |
| Thrust (lb)  | 165,000*              | 7,700 <sup>†</sup>    | 2,850 <sup>†</sup>    |
| Burning Time (sec)                                     | 148.42                | 115.35                | 35                    |
| Chamber Pressure (psia)                                |                       | 206                   | 215                   |
| Nozzle Expansion Ratio                                 |                       | 20:1                  | 26:1                  |
| Total Impulse/Weight Ratio<br>Without Payload (sec)    | 227.5*                | 204.9 <sup>†</sup>    | 230.4 <sup>†</sup>    |
| Dry Weight Without Payload (lb)                        | 7,598                 | 989                   | 60                    |
| Usable Propellant Weight (lb)                          | 96,640                | 3,241                 | 373                   |
| Payload Weight (lb) <sup>‡</sup>                       | 51.28                 | 795                   | 362                   |
| Total Takeoff Weight <sup>‡</sup><br>With Payload (lb) | 111,340               | 5,728                 | 795                   |
| Total Takeoff Weight<br>Without Payload (lb)           | 106,212               | 4,333                 | 433                   |
| Mass Ratio with Payload                                | 0.8680                | 0.6320                | 0.4692                |
| Mass Ratio without Payload <sup>‡</sup>                | 0.9098                | 0.7592                | 0.8614                |
| Fuel Outage and Trapped (lb)                           | 1,974                 | 103                   |                       |

\* At sea level

<sup>†</sup>In vacuo

<sup>‡</sup>For polar orbit

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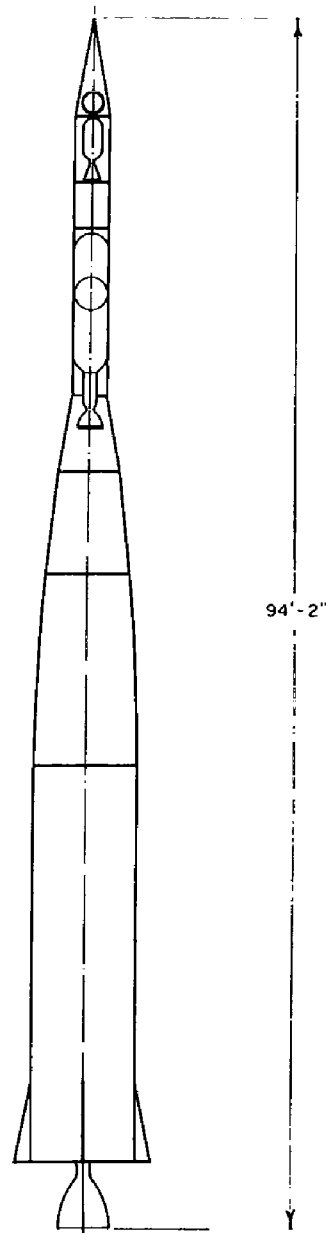


Fig. 30 - Thor-Vanguard vehicle

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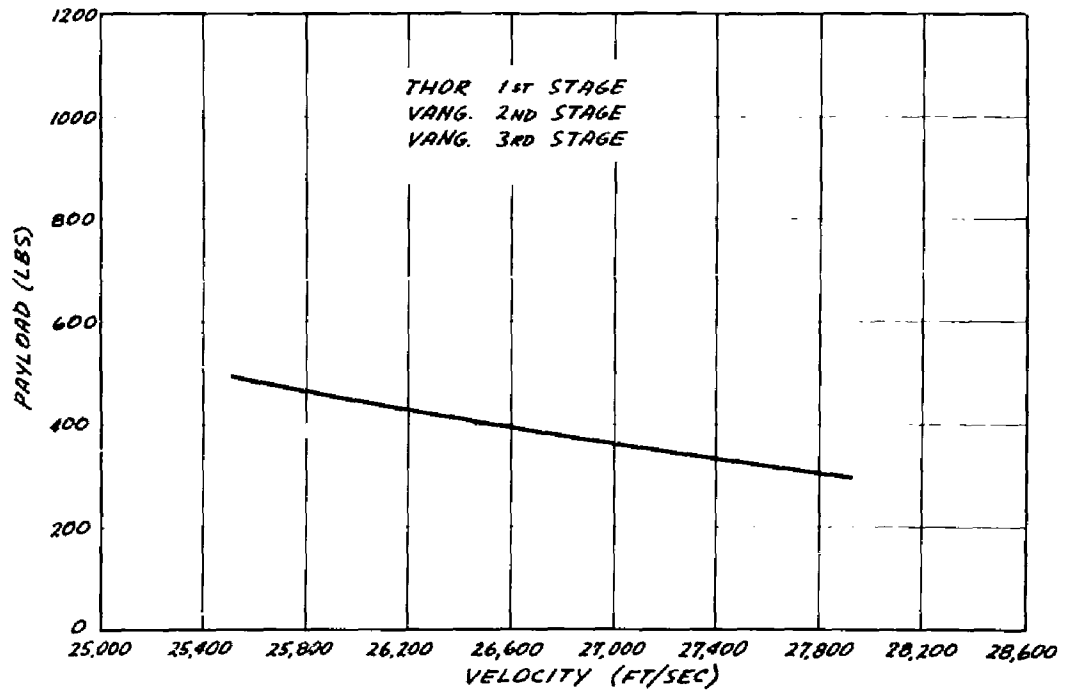


Fig. 31 - Performance of Thor-Vanguard vehicle for polar orbit

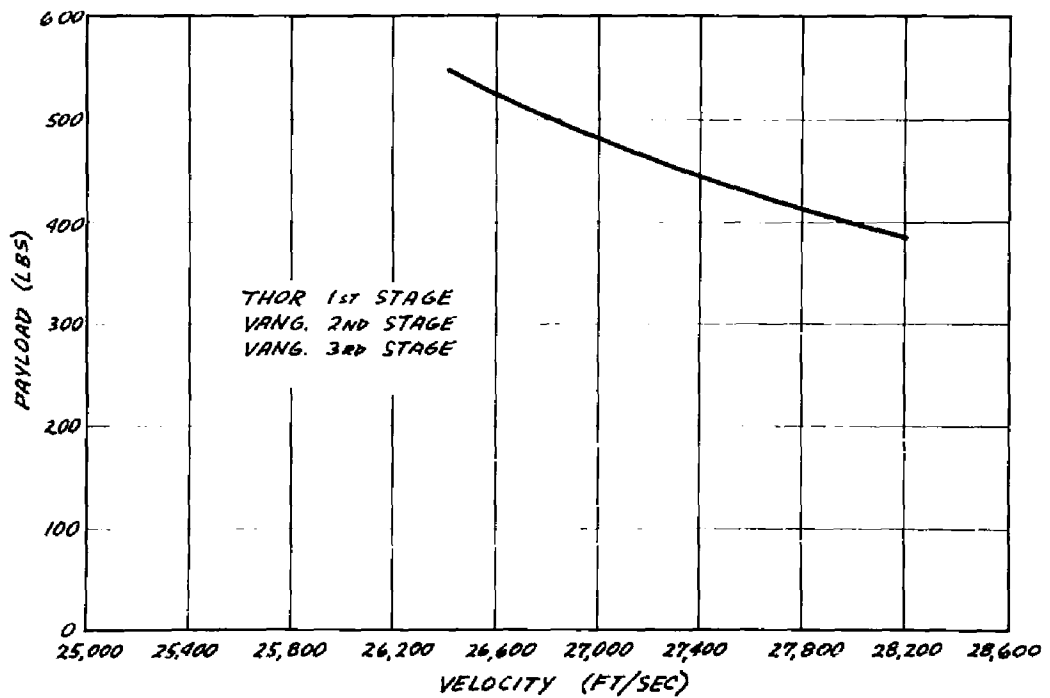


Fig. 32 - Performance of Thor-Vanguard vehicle for 35° orbit

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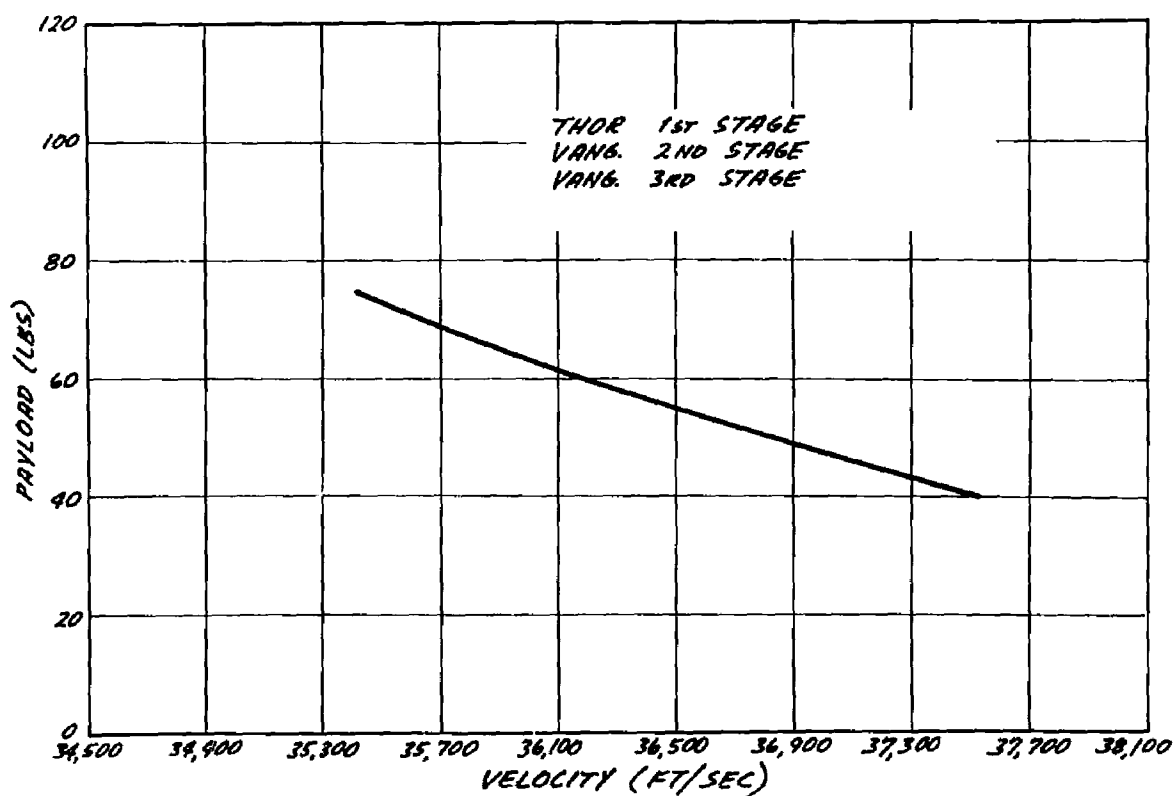


Fig. 33 - Performance of improved Thor-Vanguard three-stage vehicle for 35° orbit

SECRET-RESTRICTED DATA

4.1.4 THE IMPROVED THOR-VANGUARD THREE-STAGE  
VEHICLE (SECRET)

The Improved Thor-Vanguard three-stage vehicle is shown in Fig. 34. Its principal characteristics are shown in Table 12 and Figs. 35 and 36.

TABLE 12  
Improved Thor-Vanguard Three-Stage Vehicle Characteristics  
Projection Velocity: 27,000 fps  
Projection Altitude: 200 mi

|   | 1st Stage   | 2nd Stage  | 3rd Stage       |
|---|-------------|------------|-----------------|
| Stage Type  | Thor        | Vanguard   | 3-Vang. Cluster |
| Total Impulse (lb-sec)                              | 24,159,300* | 1,789,326† | 299,250†        |
| Specific Impulse (sec)                              | 250*        | 270†       | 265†            |
| Thrust (lb)   | 165,000*    | 7700†      | 8550†           |
| Burning Time (sec)                                  | 146.42      | 232.38     | 35              |
| Chamber Pressure (psia)                             |             | 206        | 215             |
| Nozzle Expansion Ratio                              |             | 20:1       | 26:1            |
| Total Impulse/Weight Ratio<br>Without Payload (sec) | 227.5*      | 225.2†     | 209.6†          |
| Dry Weight Without Payload (lb)                     | 7598        | 1215       | 309             |
| Usable Propellant Weight (lb)                       | 96,640      | 5530       | 1119            |
| Payload Weight (lb)‡                                | 9972        | 2028       | 600             |
| Total Takeoff Weight<br>With Payload (lb)‡          | 116,184     | 9972       | 2028            |
| Total Takeoff Weight Without<br>Payload (lb)        | 106,212     | 7944       | 1428            |
| Mass Ratio With Payload‡                            | 0.8318      | 0.6548     | 0.5518          |
| Mass Ratio Without Payload                          | 0.9098      | 0.8342     | 0.7836          |
| Fuel Outage and Trapped (lb)                        | 1.974       | 199        |                 |

\* At sea level

† In vacuo

‡ For polar orbit

SECRET-RESTRICTED DATA

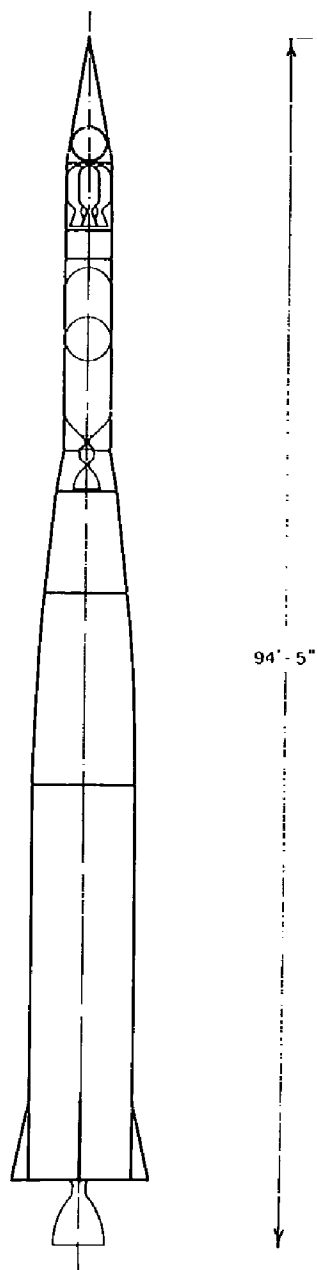


Fig. 34 - Improved Thor-Vanguard  
three-stage vehicle

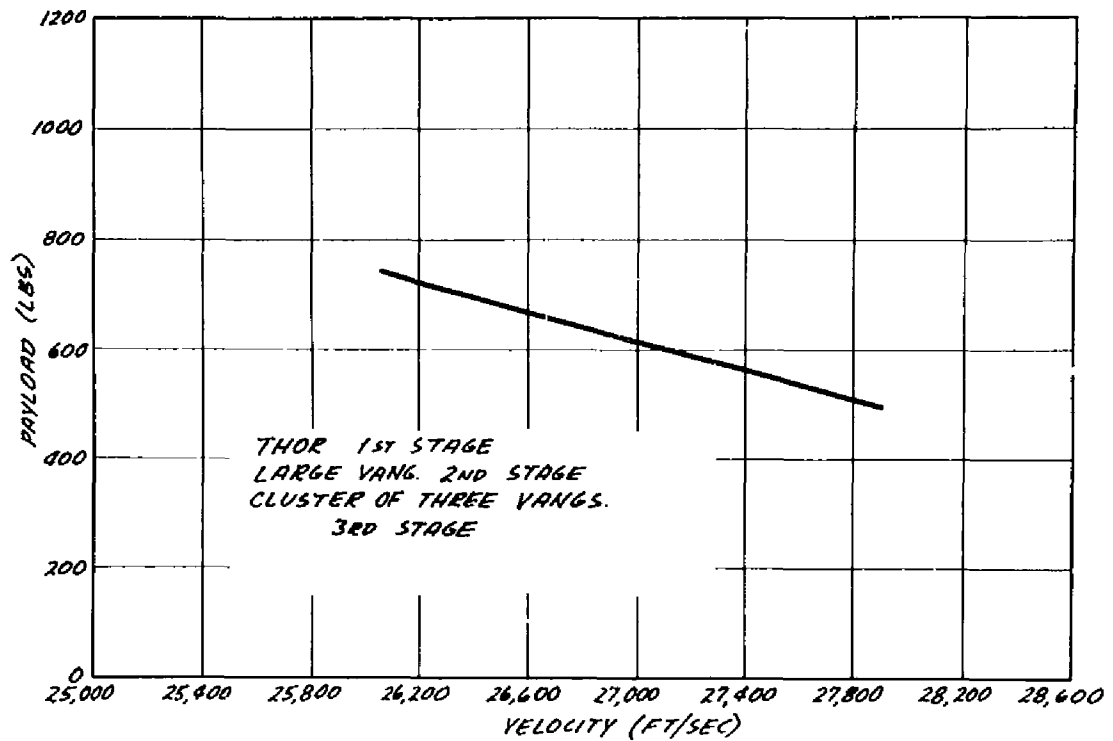


Fig. 35 - Performance of improved Thor-Vanguard three-stage vehicle for polar orbit

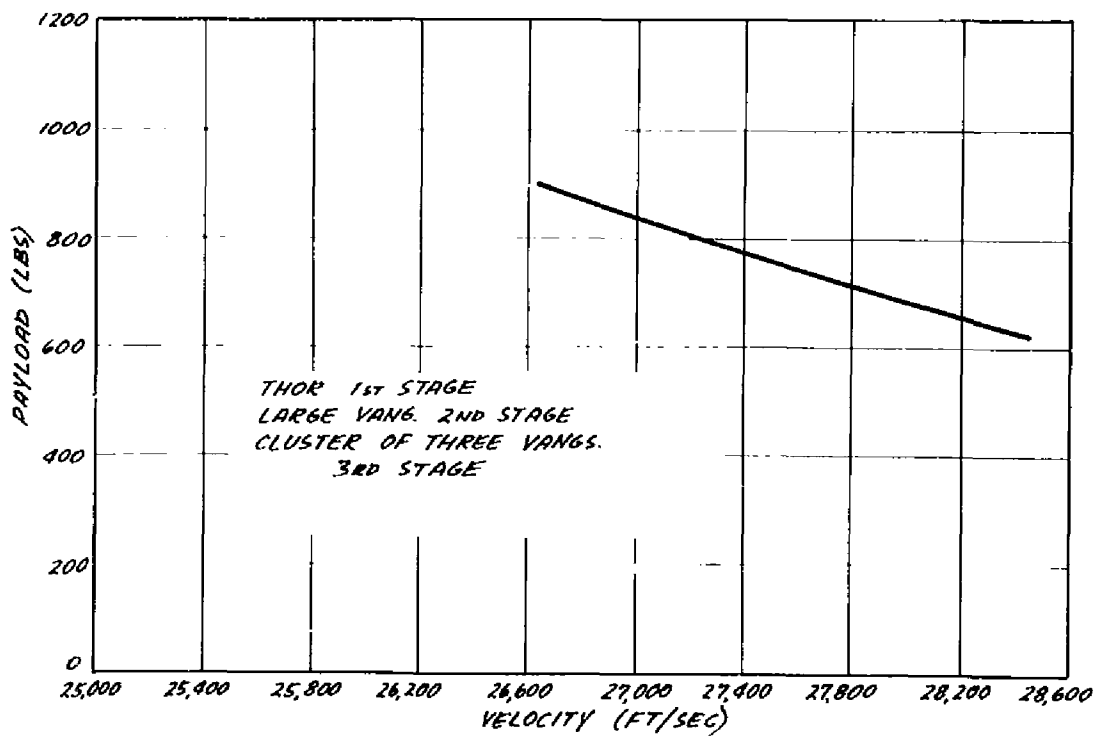


Fig. 36 - Performance of improved Thor-Vanguard three-stage vehicle for 35° orbit

SECRET-RESTRICTED DATA

4.1.5 THE IMPROVED THOR-VANGUARD FOUR-STAGE VEHICLE (SECRET)

The improved Thor-Vanguard four-stage vehicle is shown in Fig. 37. Its principal characteristics are shown in Table 13 and in Figs. 38, 39, and 40.

TABLE 13  
Improved Thor-Vanguard Four-Stage Vehicle Characteristics  
Projection Velocity: 27,000 fps  
Projection Altitude: 200 mi

|   | <u>1st Stage</u> | <u>2nd Stage</u> | <u>3rd Stage</u> | <u>4th Stage</u> |
|---|------------------|------------------|------------------|------------------|
| Stage Type  | Thor             | Vanguard         | 3 Vang. Cluster  | 1 Vanguard       |
| Total Impulse (lb-sec)                              | 24,159,300*      | 1,646,260†       | 299,250 †        | 99,750 †         |
| Specific Impulse (sec)                              | 250*             | 270†             | 265 †            | 265 †            |
| Thrust (lb)   | 165,000*         | 7,700 †          | 8,550†           | 2,850 †          |
| Burning Time (sec)                                  | 146.42           | 213.8            | 35               | 35               |
| Chamber Pressure (psia)                             |                  | 206              | 215              | 215              |
| Nozzle Expansion Ratio                              |                  | 20:1             | 26:1             | 26:1             |
| Total Impulse/Weight Ratio<br>Without Payload (sec) | 227.5*           | 223.4†           | 203.4 †          | 230.4 †          |
| Dry Weight Without Payload (lb)                     | 7,598            | 1,177            | 352              | 60               |
| Usable Propellant Weight (lb)                       | 96,640           | 5,008            | 1,119            | 373              |
| Payload Weight (lb) ‡                               | 9,972            | 2,604            | 1,133            | 700              |
| Total Takeoff Weight<br>With Payload (lb) ‡         | 116,184          | 9,972            | 2,604            | 1,133            |
| Total Takeoff Weight<br>Without Payload (lb)        | 106,212          | 7,368            | 1,471            | 433              |
| Mass Ratio With Payload ‡                           | 0.8318           | 0.6025           | 0.4297           | 0.3172           |
| Mass Ratio Without Payload                          | 0.9098           | 0.8275           | 0.7592           | 0.8614           |
| Fuel Outage and Trapped (lb)                        | 1,974            | 183              |                  |                  |

\* At sea level

† In vacuo

‡ For polar orbit

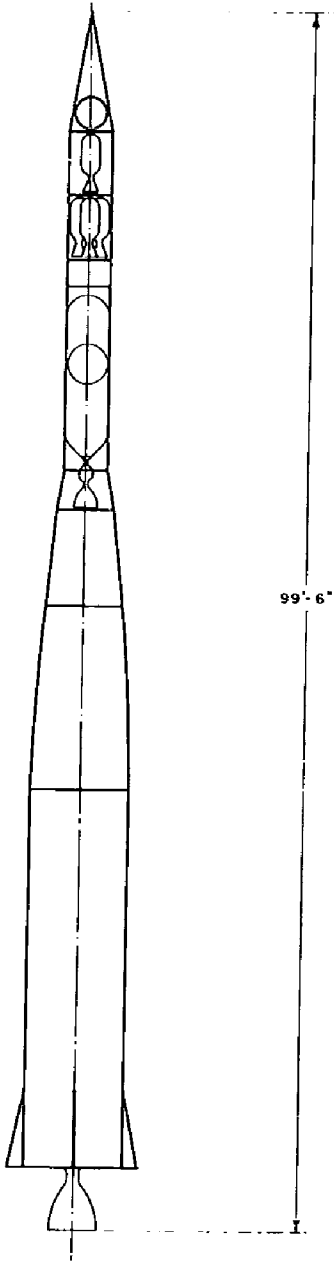


Fig. 37 - Improved Thor-Vanguard  
four-stage vehicle

SECRET-RESTRICTED DATA

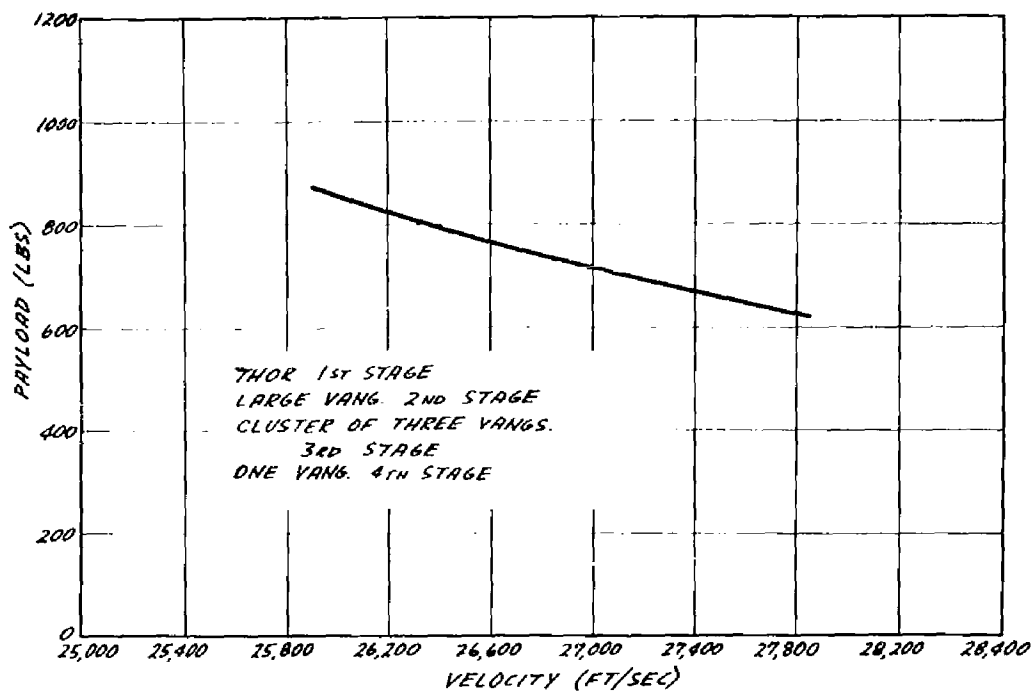


Fig. 38 - Performance of improved Thor-Vanguard four-stage vehicle for polar orbit

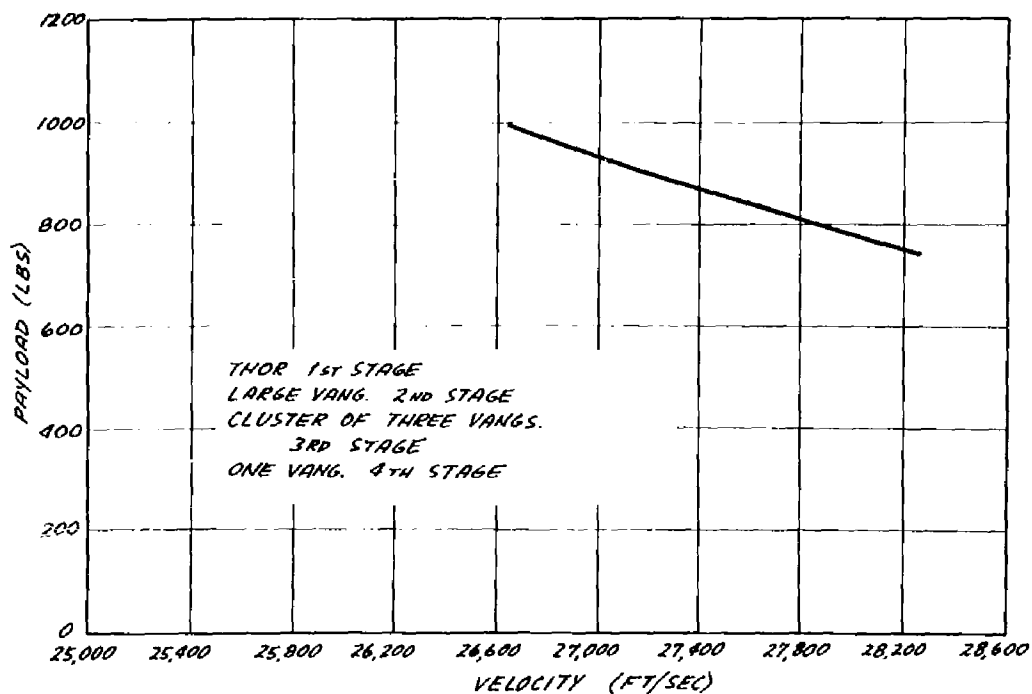


Fig. 39 - Performance of improved Thor-Vanguard four-stage vehicle for 35° orbit

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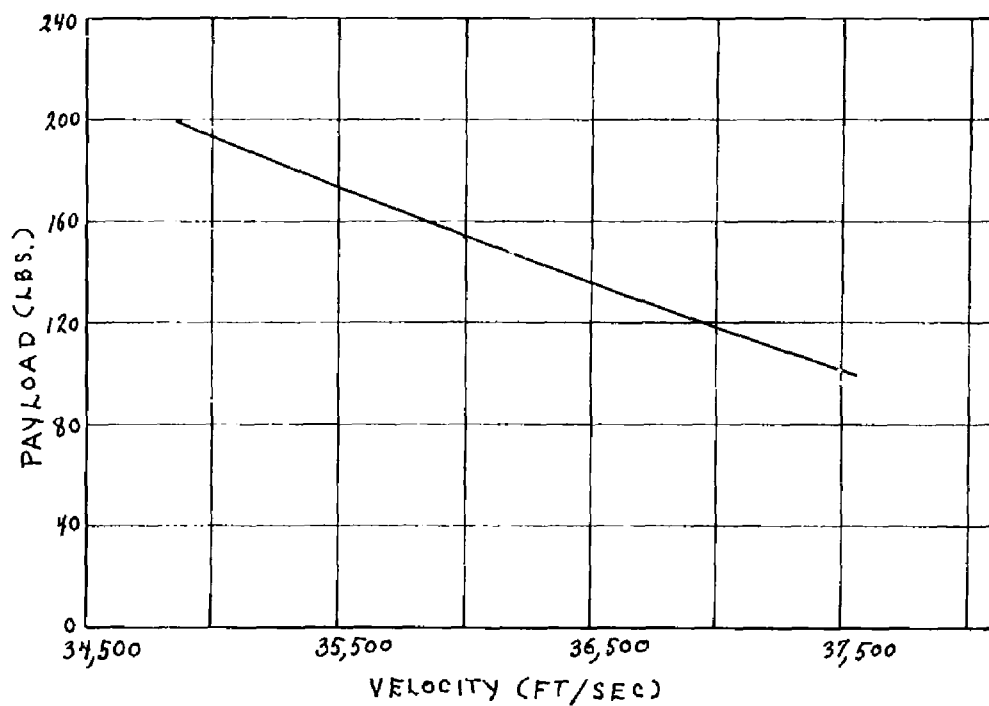


Fig. 40 - Performance of improved Thor-Vanguard four-stage vehicle for 35° orbit

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#### 4.2<sup>M</sup>SATELLITE LAUNCHING<sup>M</sup>FACILITIES

##### 4.2.1 CAPE CANAVERAL (SECRET)

The Thor and Vanguard launching facilities at Cape Canaveral are presently arranged in a fashion which would greatly facilitate the launching of a Thor-Vanguard combination. Thor and Vanguard now share the same blockhouse and, further, there exists a vacant space in the AFMTC plan for a pad which could be used to launch the Thor-Vanguard combinations. This is illustrated in Fig. 41

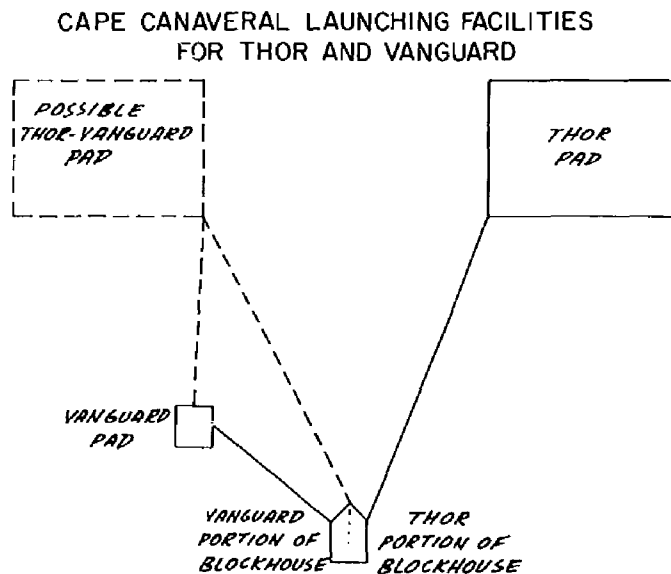


Fig. 41

##### 4.2.2 CAMP COOKE (SECRET)

The West Coast Satellite and Missile Launching Facility at Camp Cooke will be the responsibility of the Navy as executive agent. Certain operations at this establishment, including the Thor launching facility, will be conducted by the Air Force. Launching facilities for the Thor are now planned for this range. It is estimated that they will be readied on a time scale which is compatible with those of the proposed programs discussed in this report.

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# memorandum

8001/005

DATE: 15 February 2001

REPLY TO

ATTN OF: Code 8001

SUBJ: DECLASSIFICATION OF NRL REPORT 5097

TO: Code 1221.1 *ts 2/21/01*

ENCL: (1) DOE ltr ref 00SA20C000111-IB of 13 Sep 00

1. I am requesting that NRL Report 5097 be declassified. This document has been reviewed by Codes 8000 and 7600, and the Department of Energy as documented in enclosure (1) and certified that all information is no longer classified.

2. This document should be approved for public release with unlimited distribution.

  
FRED V. HELLRICH

Copy to:  
5204 (van Keuren)



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Department of Energy  
Germantown, MD 20874-1290



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release all that can be released.  
OFFICE OF NUCLEAR AND NATIONAL SECURITY INFORMATION

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In reply refer to:  
00SA20C000111-IB

Captain Douglas H. Rau  
Commanding Officer  
Attn: Code 1221.1  
Department of the Navy  
Naval Research Laboratory  
4555 Overlook Avenue SW.  
Washington, D.C. 20375-5320

Dear Captain Rau:

This responds to the memorandum from Ms. Tina Smallwood dated May 25, 2000 (enclosure 1), which requested that this office review the document at enclosure 2.

We have indicated the result of our review on the first page of the document. We have determined that the document does not contain any Department of Energy classified information; accordingly, we have no objection to its declassification and release. The document, including Section 2.1.3.1, contains no Restricted Data.

Please note that we have changed the classification category of the document to National Security Information to conform with current guidance.

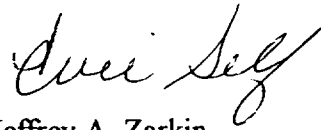


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If you have any questions, please contact Ms. Evelyn J. Self at (301) 903-8004.

Sincerely,

  
for Jeffrey A. Zarkin

Program Manager  
Statutory Reviews Program  
Document Declassification Division  
Office of Nuclear and National  
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Enclosures  
(See listing next page)

-- 1 OF 1  
-- 1 - AD NUMBER: 339967  
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